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USAAVLABS TECHNICAL REPORT 67-53

XV-5A MAINTENANCE AND SYSTEMS EVALUATION

By

R. K. Massie

July 1967

U. S. ARMY AVIATION MATERIEL LABORATORIES
FORT EUSTIS, VIRGINIA

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USAAVLABS TECHNICAL REPORT 67-53

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XV-5A

MAINTENANCE AND SYSTEMS

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By

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ABSTRACT

In the past, little formal effort has been expended by the U. S. Army in evaluating the maintenance and systems aspects of experimental aircraft.

The data compiled during this evaluation were used to determine the effectiveness of design as it applies to maintainability of the overall aircraft, its systems, and its subsystems and, in cases of deficiencies, to recommend improvements and to specify areas that require further research before derivative XV-5A-type aircraft are constructed.

Each problem area was analyzed to determine whether the discrepancies resulted from the austere research aircraft program or whether they were inherent in the lift-fan concept. Results of this study uncovered the desirable and undesirable features of 10 of the XV-5A aircraft systems.

The most undesirable feature showed that the gas-generator/lift-fan propulsion system is the focal point of high aircraft temperatures during fan flights; the high temperatures are caused by the transfer of gases from jet-engine turbines through ducts to the nose- and wing-fan turbines. The aircraft temperatures recorded during flight varied from 170° F in the electrical inverter compartment to 729° F at the inner panel of the main landing gear door. This caused a heat-soak problem, which indicated that the aircraft cooling system was not sufficient for the amount of heat being radiated. The problem was aggravated by leakage of hot gases from the ducts and by gases exhausting from the lift-fan turbines near the bottom of the fuselage.

The man-hours required for organizational maintenance amounted to 48.8 maintenance man-hours per flight hour, which does not include unscheduled maintenance. The number of man-hours is high because of the limited accessibility of systems, the required functional testing of redundant systems (the pilot could not check all of the redundant systems during ground run-ups), and the new types of systems and components. Although high, this number of man-hours is tolerable when it is considered that tests were conducted on an aircraft that was designed and fabricated for a research program.

The only spare parts that were used excessively were the silver zinc cells for the dc battery, the landing gear brake disk liners, and the ac inverters. This indicated that the overall reliability of the aircraft systems and

components was high; however, because of limited accumulative operating time, any reliability factor would be unrealistic.

Design refinements that will be required to build the lift-fan concept into an operational model are not beyond the engineering technology available during the 1967-1971 time period.

FOREWORD

Since no historical operational information was available on the lift-fan concept, an attempt was made during the XV-5A test program to obtain information that would be helpful to groups that are working on lift-fan aircraft concepts in areas such as design, reliability and maintainability, evaluation, and logistics.

The XV-5A flight test program began on 27 January 1965 at Edwards Air Force Base, California, with two aircraft. One of the aircraft was lost in an accident on 27 April 1965. Tests were continued with the remaining aircraft, and the program was concluded on 15 November 1965.

Since it is necessary to know the effect of deficiencies early in a development program so that effective corrective action can be taken, the author was assigned full time at the test site at Edwards Air Force Base, California, to establish a system for recording as much detailed information as possible, with the manpower available, and to analyze each failure or problem as it occurred.

The information compiled from limited samples, limited component tests, and limited flight tests conducted on this research aircraft between January and November 1965 is inadequate to establish factually the requirements for the maintenance of the aircraft systems, subsystems, and components of a new-concept aircraft. However, it is believed that the information is sufficiently valuable to determine the requirements for similar aircraft and that this information could be of assistance to various groups in selecting concepts for future procurement.

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INTRODUCTION

During the XV-5A flight test program conducted between January and November 1965, the discrepancies totalled 313, of which 156 were attributed to equipment failure. Twenty-five of these failures (6 flight aborts, 2 flight terminations, and 17 flight delays) affected a scheduled flight mission. It was believed that information pertaining to these failures would be valuable for evaluation purposes.

To establish a basis for formulating opinions and judgments in this evaluation study, the analysis considered the lift-fan concept of an operational aircraft that would be flown at a ratio of 1 hour in fan (hover) mode to 10 hours in jet (conventional) mode.

Data for the study were compiled from the following sources: reports of pilots, design engineers, and maintenance technicians; equipment failure reports; operating time logs; and maintenance operations inspection records (such as those for time-compliance inspections, functional tests, and operational inspection discrepancies).

The topics selected for study were reviewed and analyzed on the basis of pertinence and/or criticality with respect to future designs. The methods and procedures to be used during this study were also considered in the selection of topics.

The problems which occurred during the flight test program were considered carefully to determine whether they could recur in service or whether they were isolated instances. Following are several factors that were taken into consideration:

1. If failures occurred at all during a relatively short test period, it was assumed that they would be repeated frequently if the aircraft were in service. In general, the flight time accumulated was small in comparison to the expected time between overhauls when an aircraft is actually in service.

2. Failures that were predominately dependent upon operating time or upon the number of cycles, such as fatigue failures, may not have been discovered, because there was insufficient operating time for such failures to develop.

3. The aircraft flew from the same base and was maintained by the same crew, so the effect of operating many aircraft from many bases with many crews could only be estimated.

DESCRIPTION OF AIRCRAFT

The XV-5A is a midwing, turbojet-powered research aircraft. (Figure 1 shows three views of the XV-5A, Figure 2 is a cutaway drawing of the aircraft, and Figure 3 shows the airframe and structural arrangement.) The propulsion system consists of two J-85 engines, two X353-5B wing fans, and one X376 nose pitch fan.

The aircraft has the following capabilities: conventional wing-supported flight at high subsonic speeds; vertical takeoff and landing (VTOL) and short takeoff and landing (STOL) in the fan-supported flight mode; transition from hovering flight to high horizontal speed flight and back to hovering flight; and conventional takeoff and landing (CTOL). During wing-supported flight, conventional aerodynamic control surfaces are utilized. During fan-supported flight, control is accomplished through modulation of the airflow through the fans.

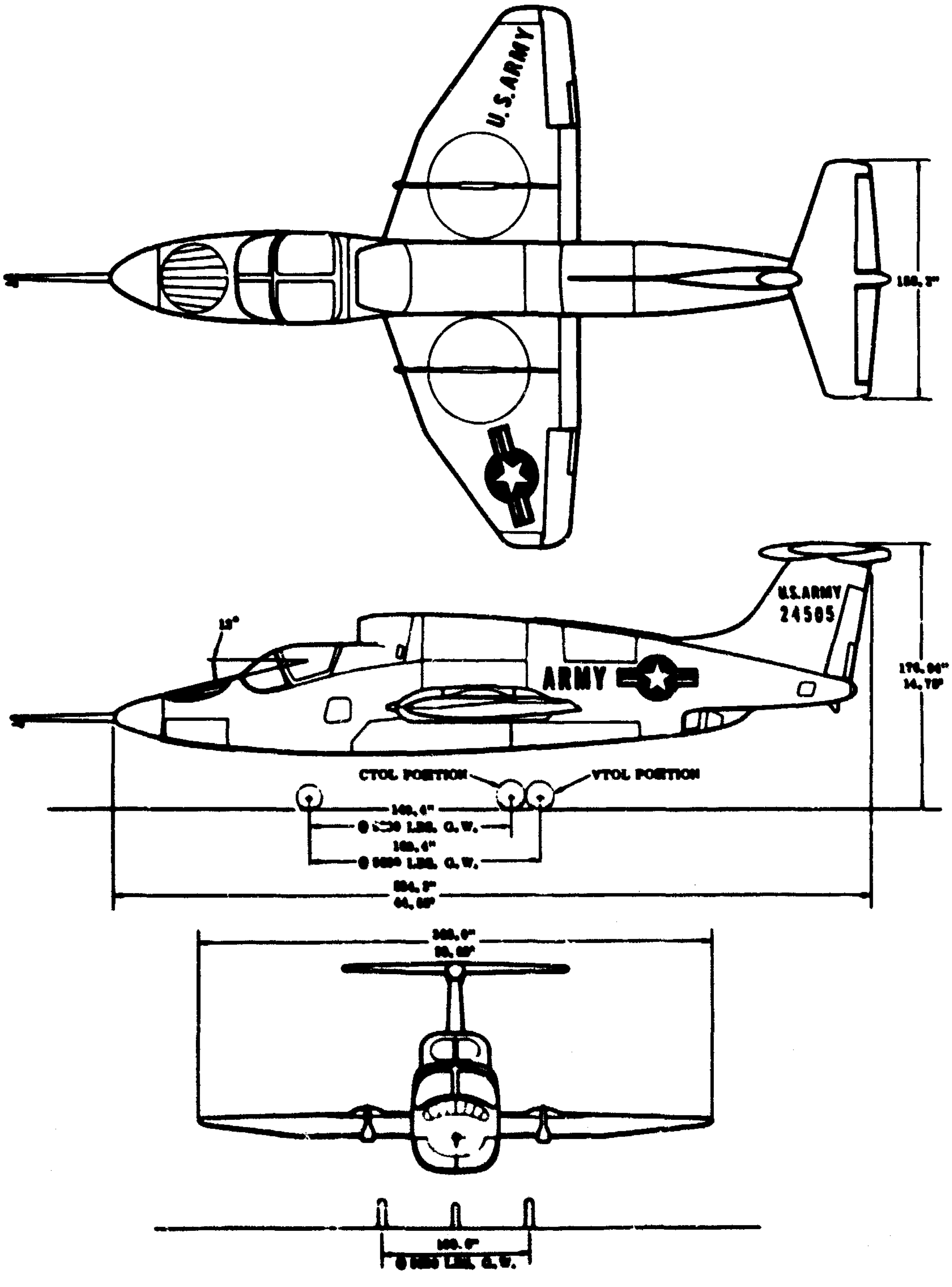


Figure 1. Three Views of the XV-5A.

1. PITOT MAST
2. FIBER GLASS NOSE CONE
3. X376 PITCH FAN
4. NOSE-FAN THRUST CONTROL DOOR
5. NOSE-FAN INLET CLOSURE DOORS
6. WINDSHIELD
7. NOSE-FAN SUPPLY DUCT
8. RUDDER PEDALS
9. INSTRUMENT PANEL
10. CONVENTIONAL CONTROL STICK
11. OBSERVER'S EJECTION SEAT
12. NOSE LANDING GEAR
13. THROTTLE QUADRANT
14. PILOT'S EJECTION SEAT
15. COLLECTIVE LIFT STICK
16. HYDRAULIC EQUIPMENT COMPARTMENT
17. SINGLE SPLIT ENGINE INLET DUCT
18. ELECTRICAL EQUIPMENT COMPARTMENT
19. HYDRAULIC PUMP
20. FWD MAIN FUEL TANK
21. GENERATOR
22. RIGHT WING
23. J-85 GAS GENERATOR
24. RIGHT-HAND AILERON
25. CROSSOVER DUCT
26. WING-FAN LOUVER ACTUATORS
27. DIVERTER VALVE
28. WING-FAN INLET CLOSURE DOORS
29. X353-5B LIFT FAN
30. ENGINE TAIL PIPE
31. TWO-POSITION MAIN LANDING GEAR
32. LEFT WING
33. LEFT-HAND AILERON
34. LEFT-HAND WING FLAP
35. LEFT-HAND THRUST SPOILER
36. EXTERNAL LONGERON
37. VERTICAL FIN
38. FULL MOVABLE HORIZONTAL STABILIZER
39. ANTISPIN AND DRAG CHUTE COMPARTMENT
40. RUDDER
41. ELEVATORS

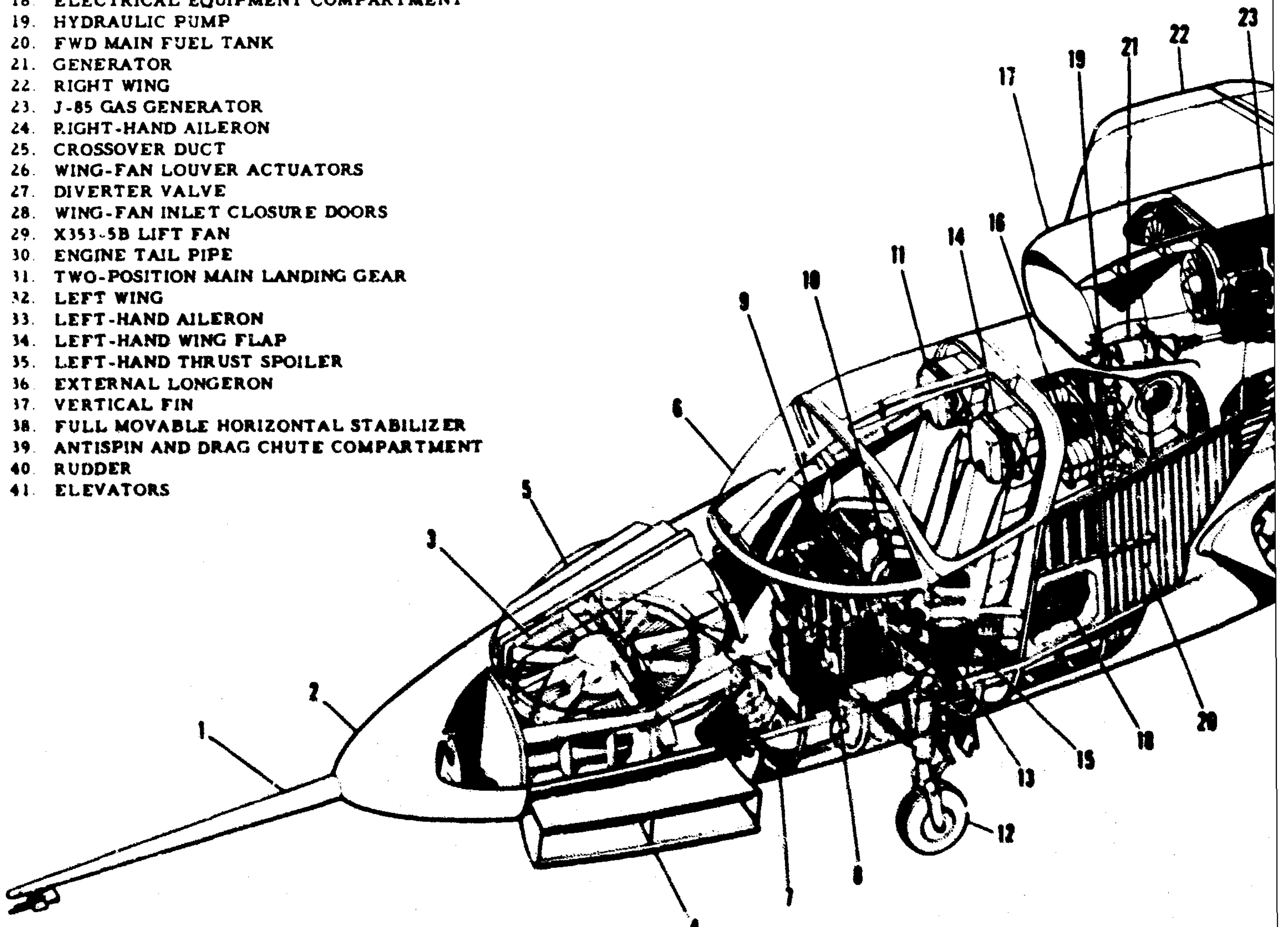
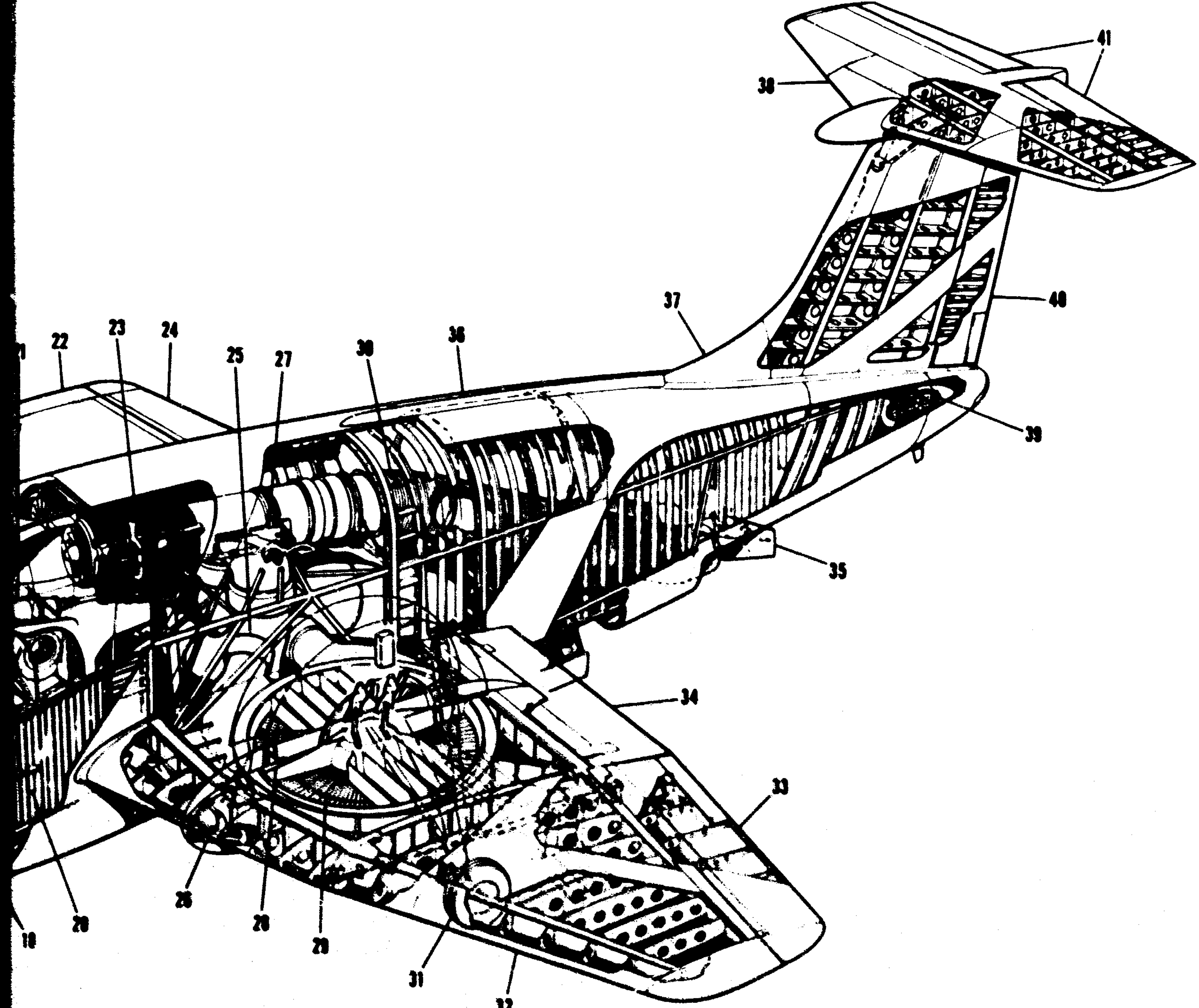


Figure 2. Aircraft Cutaway Drawing.



B

1. NOSE CONE
2. FORWARD BULKHEAD, PITCH FAN
3. FORWARD FUSELAGE SECTION
4. AFT BULKHEAD, PITCH FAN
5. WINDSHIELD
6. CANOPY
7. FRONT SPAR BULKHEAD
8. CANTED BULKHEAD, FORWARD FUSELAGE
9. LEFT-HAND LOWER LONGERON, FORWARD FUSELAGE
10. LEFT-HAND UPPER LONGERON, FORWARD FUSELAGE
11. RIGHT-HAND WING
12. RIGHT-HAND AILERON
13. RIGHT-HAND FLAP
14. RIGHT-HAND ENGINE MASTER MOUNTS
15. LEFT-HAND ENGINE MASTER MOUNTS
16. CENTER FUSELAGE SPACE FRAME
17. REAR SPAR BULKHEAD
18. AFT FUSELAGE SECTION
19. EXTERNAL LONGERON
20. LEFT-HAND UPPER LONGERON, AFT FUSELAGE
21. VERTICAL STABILIZER LEADING EDGE FAIRING
22. VERTICAL STABILIZER
23. VERTICAL STABILIZER REAR SPAR
24. VERTICAL STABILIZER CENTER SPAR
25. TAIL CONE
26. VERTICAL STABILIZER REAR SPAR BULKHEAD
27. VERTICAL STABILIZER FORWARD SPAR
28. VERTICAL STABILIZER RIB
29. VERTICAL STABILIZER CENTER SPAR BULKHEAD
30. VERTICAL STABILIZER FORWARD SPAR BULKHEAD
31. TAIL PIPE AFT BULKHEAD
32. TAIL PIPE EXHAUST FAIRING
33. LEFT-HAND LOWER LONGERON, AFT FUSELAGE
34. REAR WING SPAR FUSELAGE ATTACH STRUCTURE
35. RUDDER
36. RUDDER TRIM TAB
37. HORIZONTAL STABILIZER
38. HORIZONTAL STABILIZER FORWARD SPAR
39. HORIZONTAL STABILIZER TIP
40. HORIZONTAL STABILIZER CENTER SPAR
41. HORIZONTAL STABILIZER REAR SPAR
42. HORIZONTAL STABILIZER RIB
43. ELEVATORS
44. ELEVATOR RIB
45. LEFT-HAND FLAP
46. LEFT-HAND AILERON
47. LEFT-HAND AILERON AFT SPAR
48. LEFT-HAND AILERON RIB
49. LEFT-HAND AILERON FRONT SPAR
50. LEFT-HAND WING AFT SPAR
51. CAP RIB
52. LEFT-HAND WING TIP
53. LEFT-HAND OUTBOARD WING PANEL
54. LEADING EDGE FAIRING
55. LEFT-HAND WING FRONT SPAR
56. INBOARD WING PANEL
57. NOSE-FAN-PITCH CONTROL DOORS
58. MAIN LANDING GEAR
59. MAIN LANDING GEAR DOORS
60. NOSE LANDING GEAR
61. AILERON TRIM TAB
62. CENTER FUSELAGE UPPER ACCESS COVER
63. CENTER FUSELAGE LOWER ACCESS COVER
64. CENTER FUSELAGE SIDE ACCESS COVER
65. STRUT

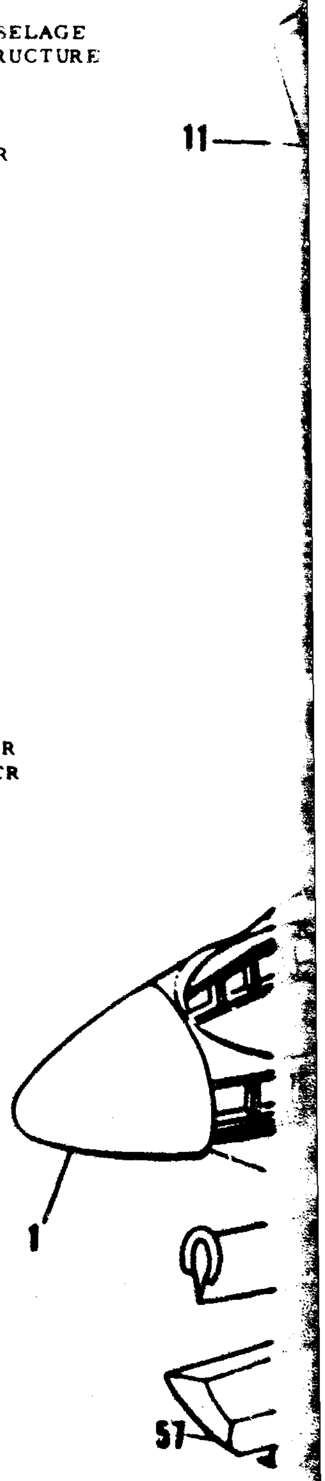


Figure 3. Airframe Structural Arrangement.

LONGERON, AFT FUSELAGE
ISELAGE ATTACH STRUCTURE

.IZER
.IZER FORWARD SPAR
.IZER TIP
.IZER CENTER SPAR
.IZER REAR SPAR
.IZER RIB

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N AFT SPAR
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N FRONT SPAR
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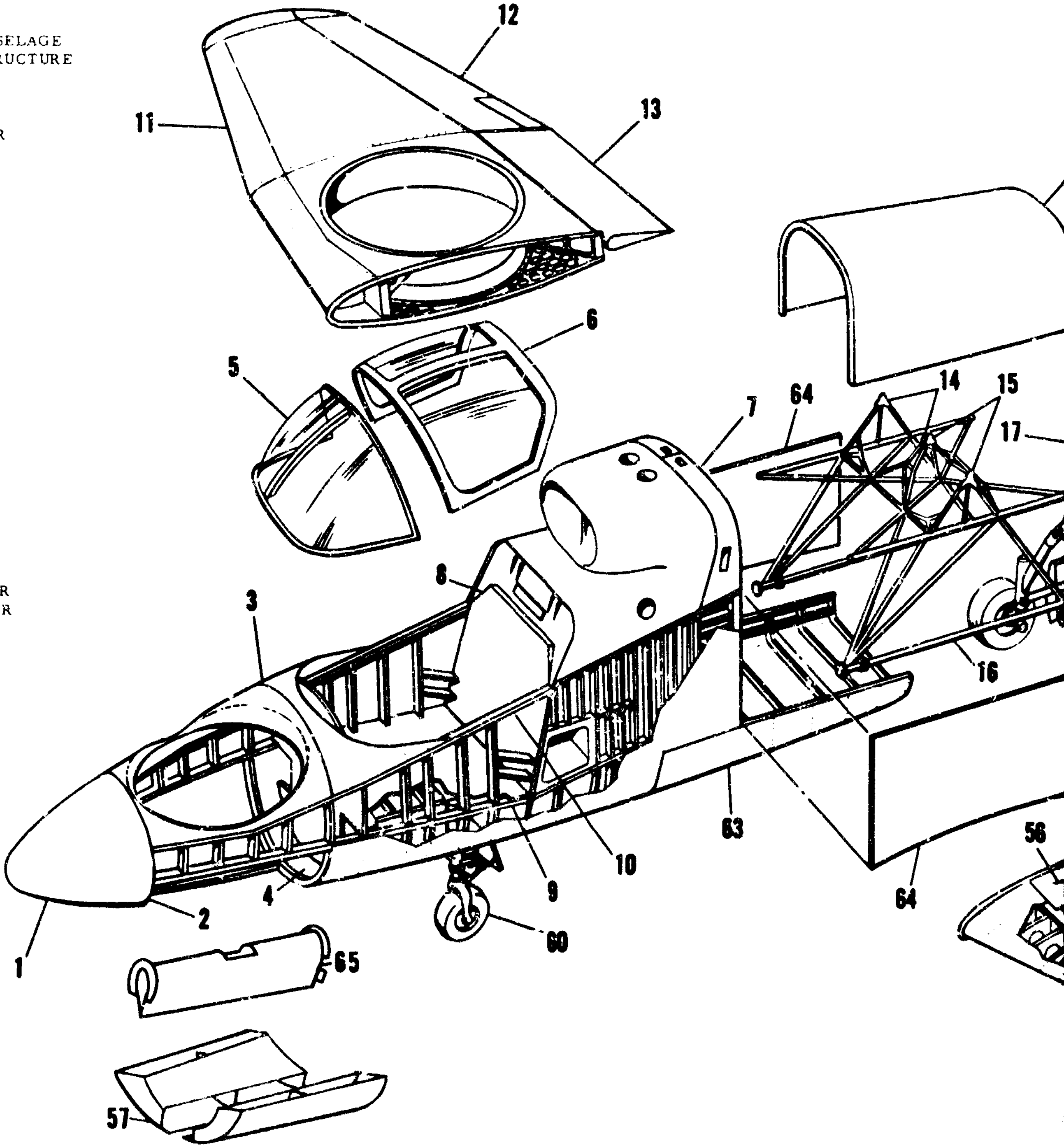
UPPER ACCESS COVER
LOWER ACCESS COVER
SIDE ACCESS COVER



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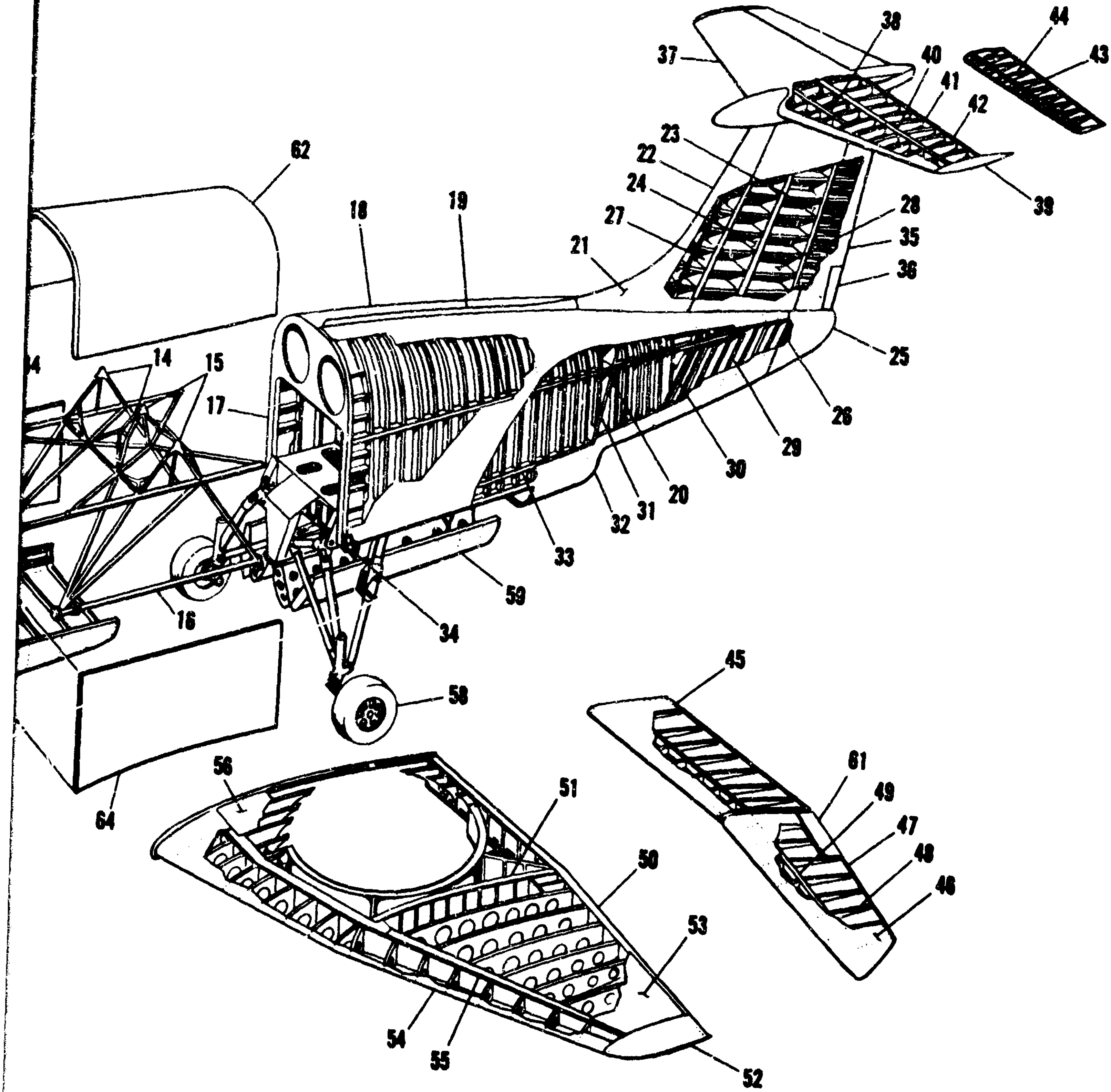

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PROCEDURES AND RESULTS

Each problem area was analyzed to determine whether the discrepancies resulted from the austere research aircraft program or whether they were inherent in the lift-fan concept. The analysis was based on 273 test missions, which included 114 ground tests and 159 flight missions, of which 128 flights and 19:40 hours of fan time were logged. A total of 110:35 test hours was accrued from 59:50 hours of flight tests and 50:45 hours of ground tests. Operating times and discrepancy rates were computed on a weekly basis, and the trends were analyzed to support recommendations for improving the program.

Following are the results and findings of the analysis:

1. Many of the discrepancies were induced by the maintenance technicians while they were still learning, and others were attributed to the aircraft's being debugged. A large number of these discrepancies had occurred at the beginning of the flight test program, when very few missions had been flown. At the conclusion of a short training program, and after the inspection requirements were updated, the ratio between the number of discrepancies and flights improved; 3 discrepancies per flight dropped to 1.64 discrepancies per flight.

2. Extending the aircraft functional checkout effective period from 24 hours to 48 hours resulted in sufficient man-hours saved to permit a three- instead of a two-shift operation without an increase in manpower; this increased the average aircraft availability rate from .70 to .86 flight per working day.

3. Of the 313 documented discrepancies, 87 percent were discovered when the aircraft was on the ground.

4. Because of component failures, excessive man-hours were required for restoring the aircraft.

5. A list of recommended improvements was compiled; the compilation was based on an analysis of the comments of pilots, engineers, and maintenance technicians.

6. When the XV-5A is operated in the jet mode, it can be compared to a present-day jet-type aircraft, such as the F-5. When it is operated in the fan mode, a comparison can not be made, since there are no other aircraft with this feature.

AIRFRAME COOLING SYSTEM

SYSTEM CONFIGURATION AND OPERATION

Description of System

The gas-generator/lift-fan propulsion system is the focal point of any aircraft heating problem and of cooling system performance (see Figure 4). The cooling system consists of two generally parallel branches, separated by a vertical plane through the aircraft centerline (BL = 0); the two branches have a few ducts and plenums in common.

The cooling system is made up of upper and lower fuselage sections, which are shown separated by the cross-hatched line in Figure 5. Figures 6 through 13 show the cooling system details.

The primary motive power for each branch is supplied by two cooling air blowers. Each set of blowers is driven separately by one of the two gas generators. The blowers for both branches are housed in a common plenum and draw outside air from two fuselage ports, which supply the plenum, and from a slot formed by the cockpit canopy closure. This slot also acts as a second boundary layer bleed duct and provides for cockpit ventilation as well. The smaller blower cools the electrical generator, the hydraulic oil cooler, and the electronic compartment before dumping into the lower fuselage section. The large blower supplies cooling air to the engine compartment and/or the crossover duct compartment and wings, depending upon the mode of operation (turbojet or fan). Cooling air pumping is augmented by the tail-pipe ejector during turbojet-mode operation and by the nose- and wing-fan cooling air ejectors during fan-mode operation. The nose-fan air ejectors consist of slots cut into the nose-fan inlet louver support struts running fore and aft across the bellmouth.

During operation in the lift-fan mode, the high-velocity air flowing over the struts to the nose fan creates a low static pressure at the slots and thereby develops a differential pressure across the nose-fan compartment cooling system. The ejector does not operate in the conventional mode. The two (fore and aft) air ejectors on each wing fan are located on the fan strut running fore and aft on the fan centerline and across the bellmouth, as presented in Figure 12. During operation in the lift-fan mode, the high-velocity cold air entering the wing fan creates a low static pressure at the ejectors and thereby develops a differential pressure across the wing-fan compartment cooling system. The air ejectors are nonoperative during conventional operation.

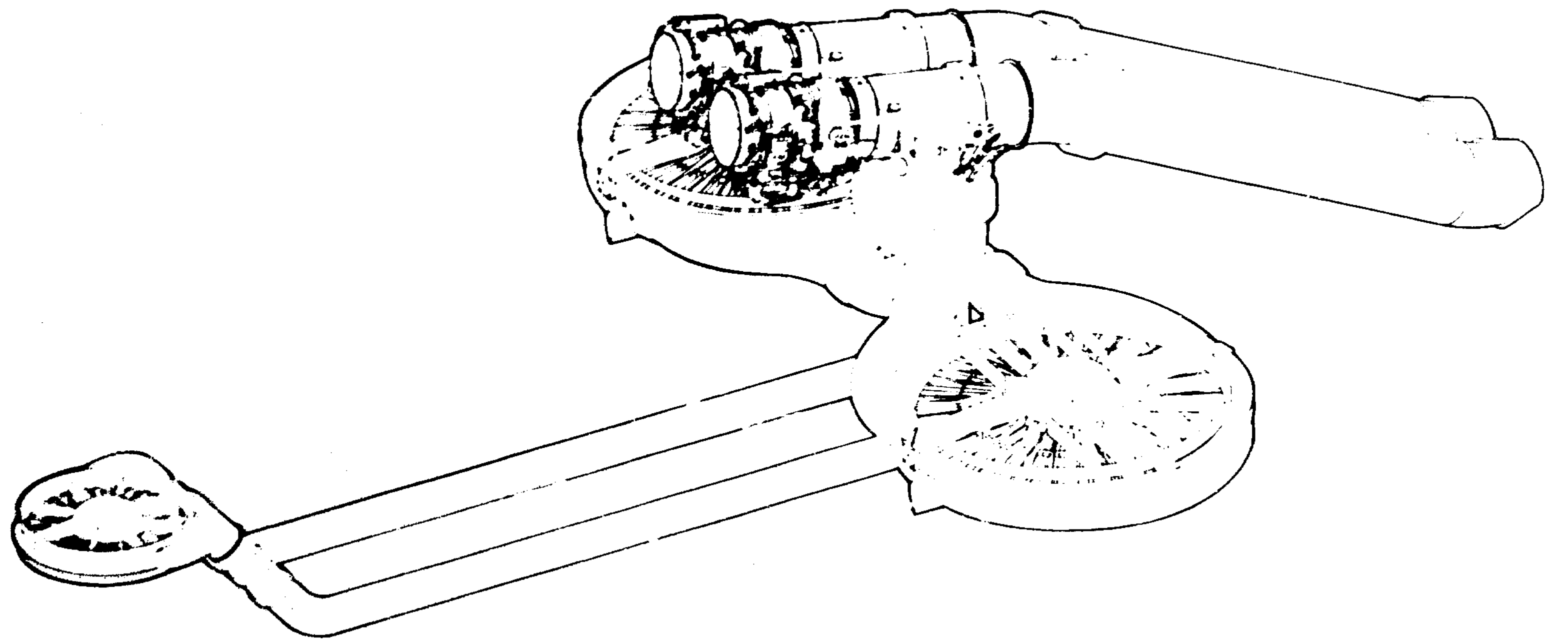

Figure 4. Propulsion Lift System.

COMPARTMENT BOUNDARIES

BOUNDARY BETWEEN UPPER & LOWER FUSELAGE

A COCKPIT
B ELECTRONIC COMPARTMENT
C HYDRAULIC COMPARTMENT
D COOLING FAN COMPARTMENT
E ENGINE INLET AND BOUNDARY LAYER BLEED SECTION
F ENGINE COMPRESSOR COMPARTMENT
G ENGINE BAY
H TAIL PIPE AND SHROUD SECTION
J AFT FUSELAGE COMPARTMENT
K AFT EQUIPMENT COMPARTMENT
L MAIN LANDING GEAR COMPARTMENT
M FLAP ACTUATOR COMPARTMENT
N CENTER FUSELAGE COMPARTMENT
O MID FORWARD LOWER FUSELAGE COMPARTMENT
P FORWARD LOWER FUSELAGE COMPARTMENT
Q NOSE-FAN COMPARTMENT
R NOSE-FAN CAVITY
S DRAG CHUTE COMPARTMENT
T TAIL SECTION

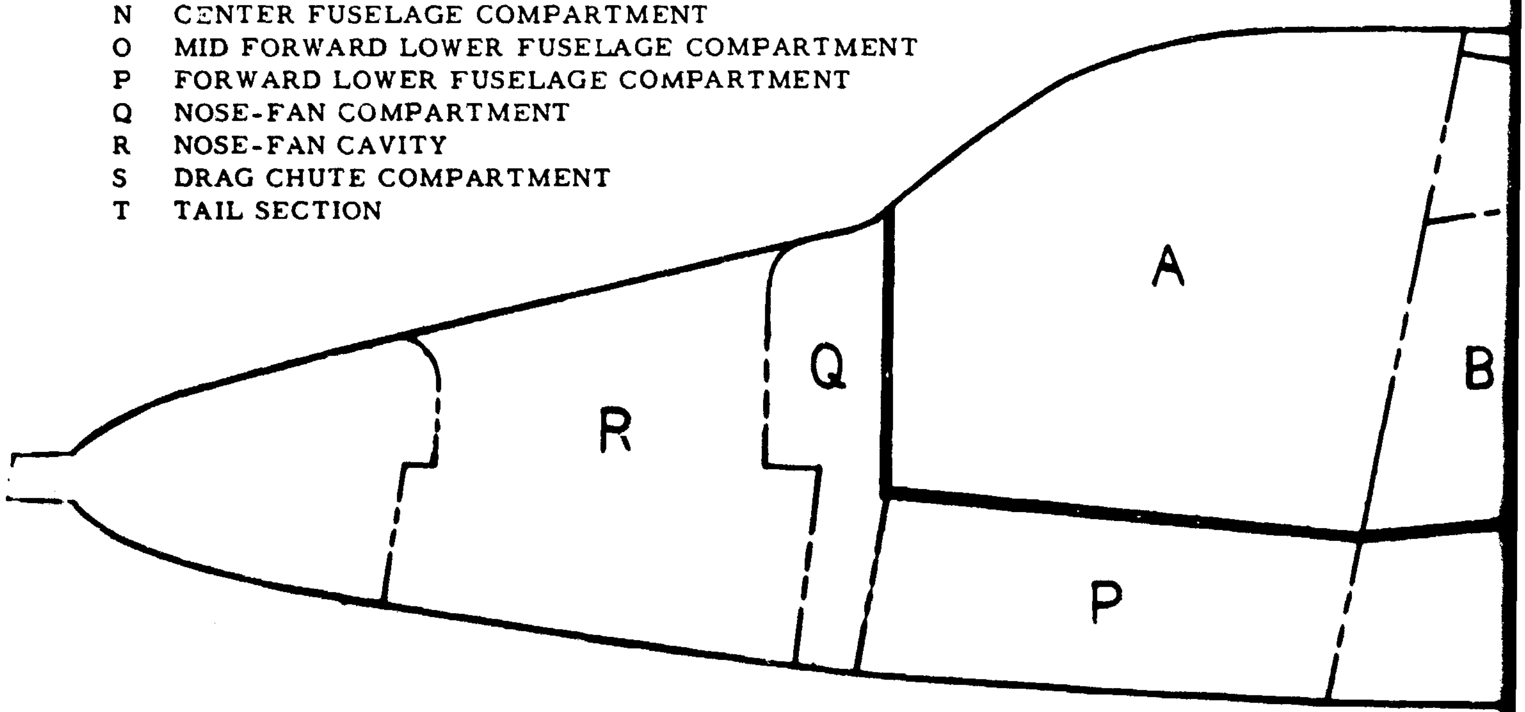

Figure 5. General Arrangement - Fuselage Section and Compartments.

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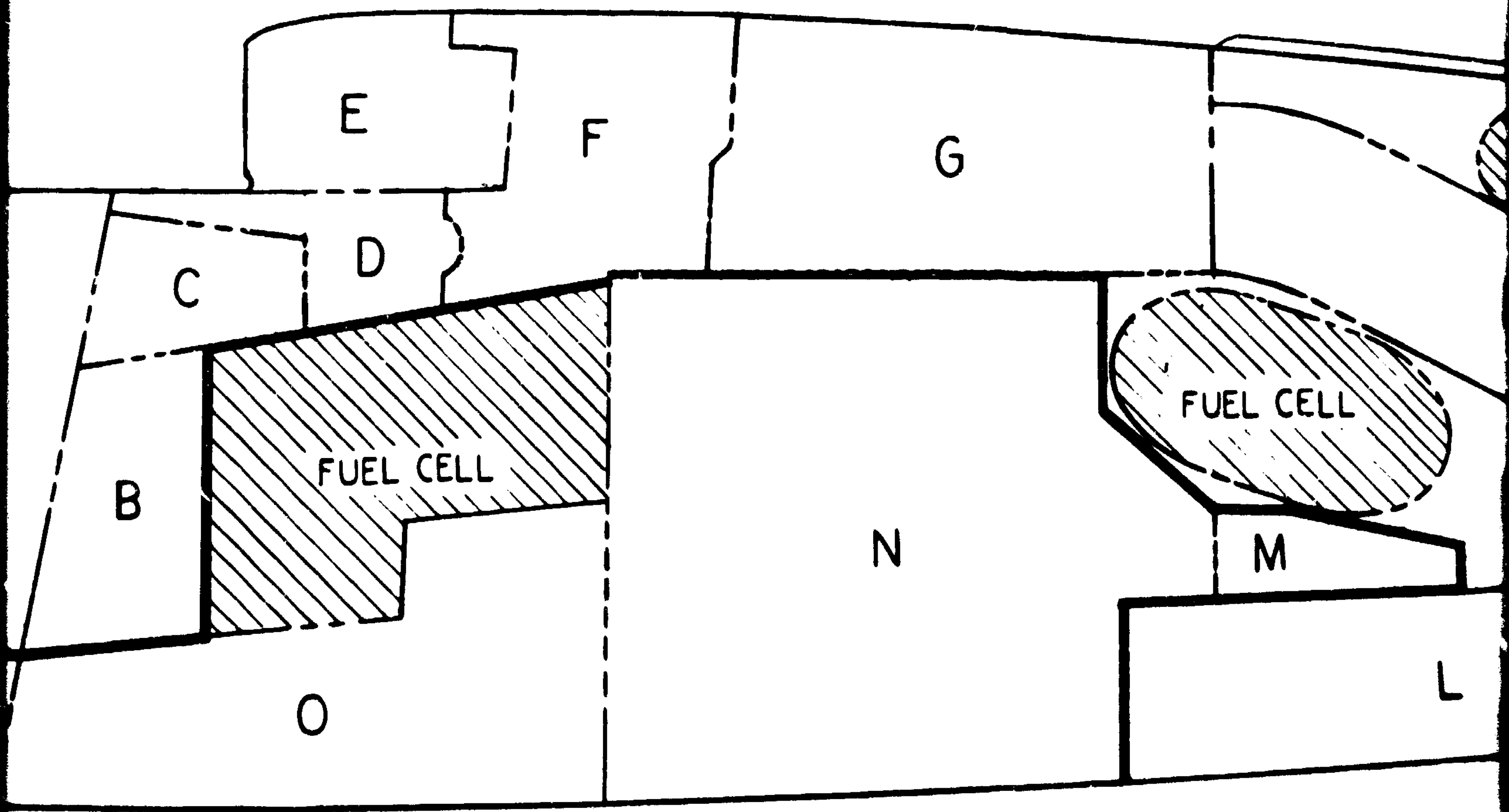

E
F
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FUEL CELL
FUEL CELL
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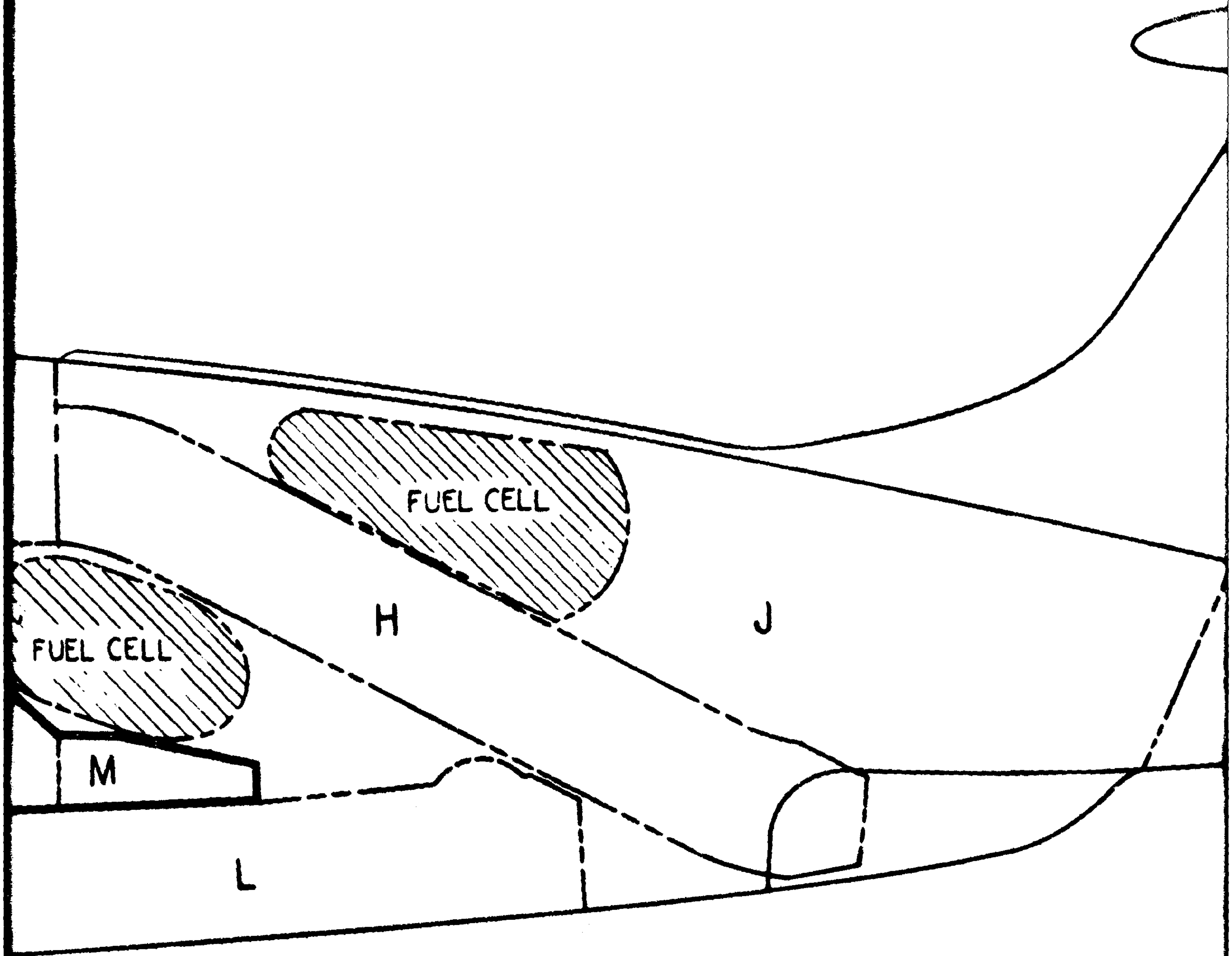

FUEL CELL
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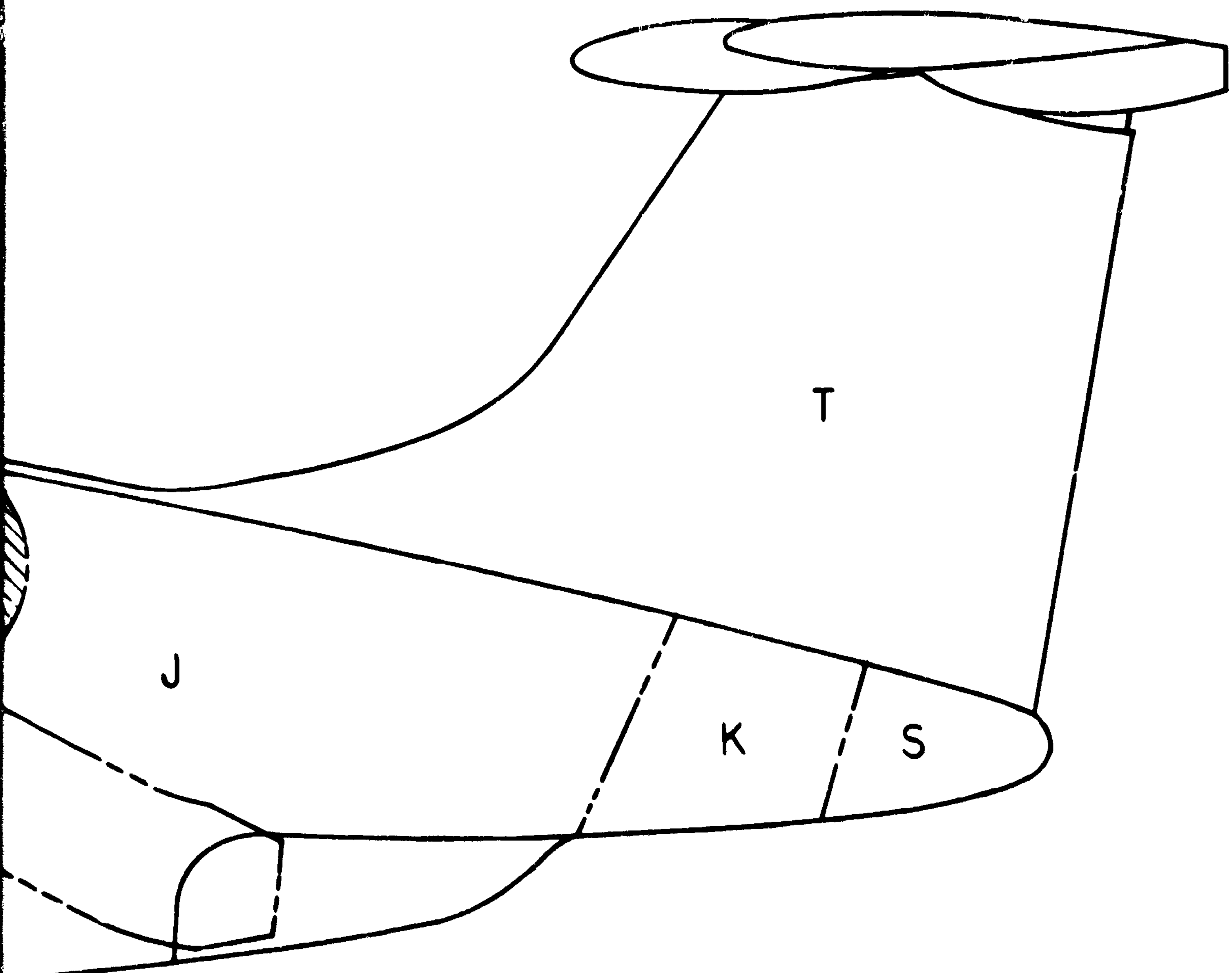

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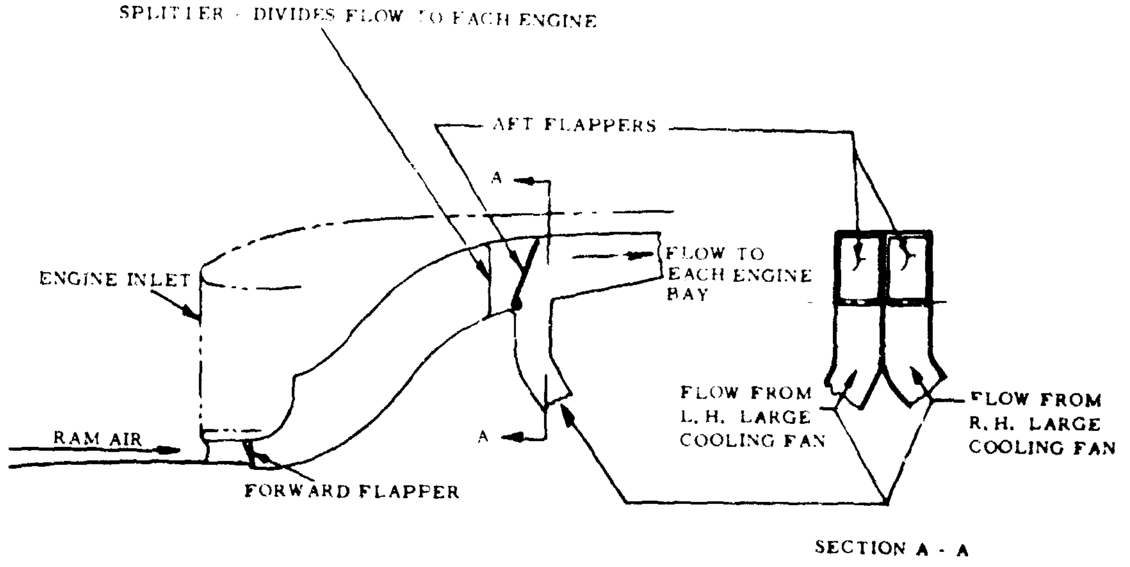


Figure 6. Boundary Layer Bleed Duct.

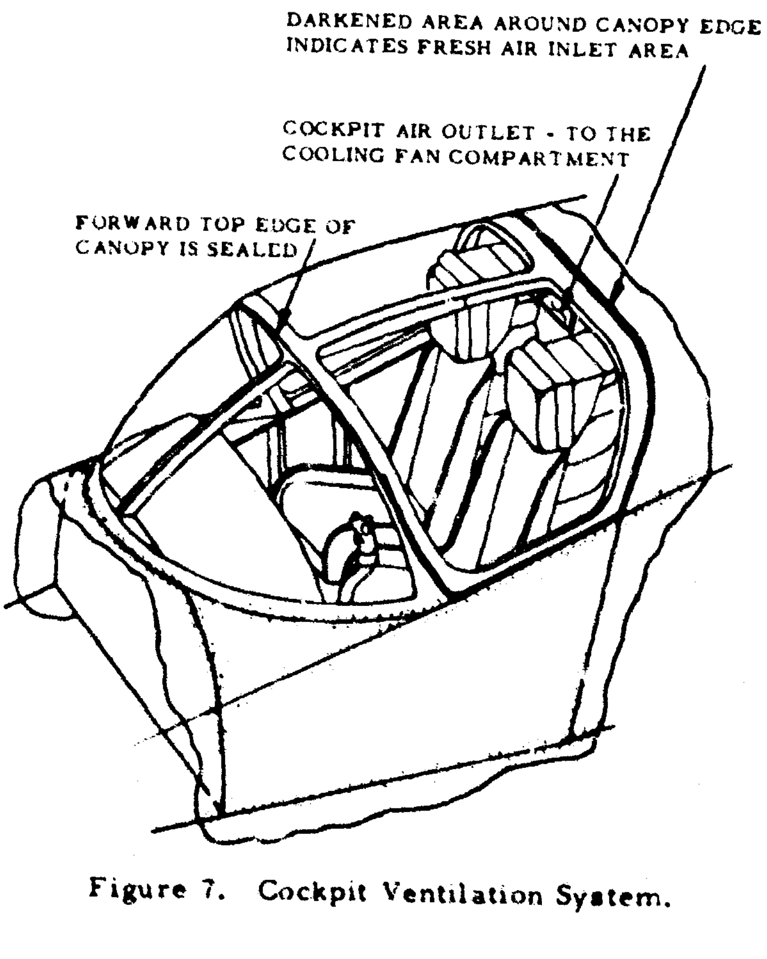


Figure 7. Cockpit Ventilation System.

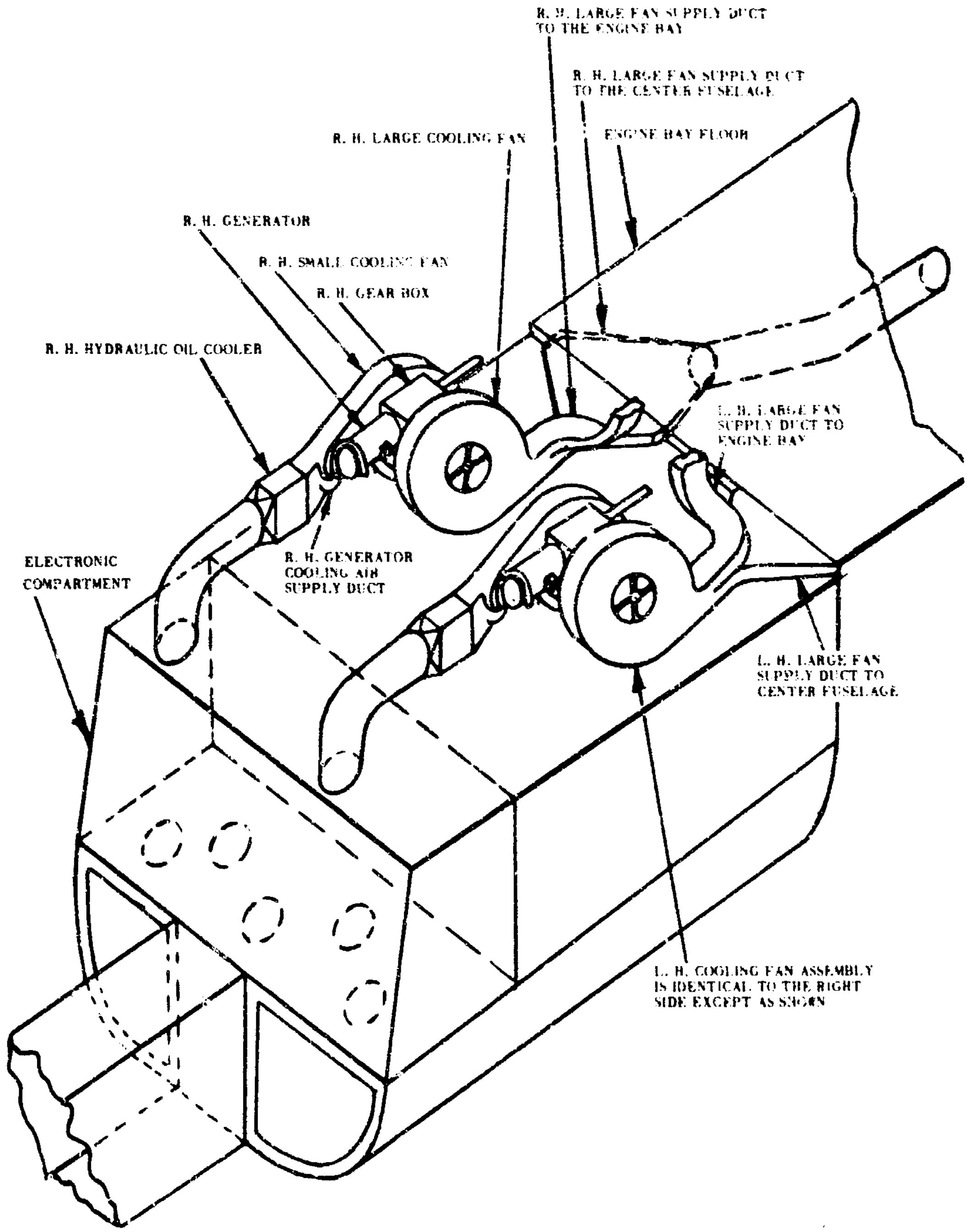


Figure 8. Cooling Fan, Hydraulic and Electronic Compartments.

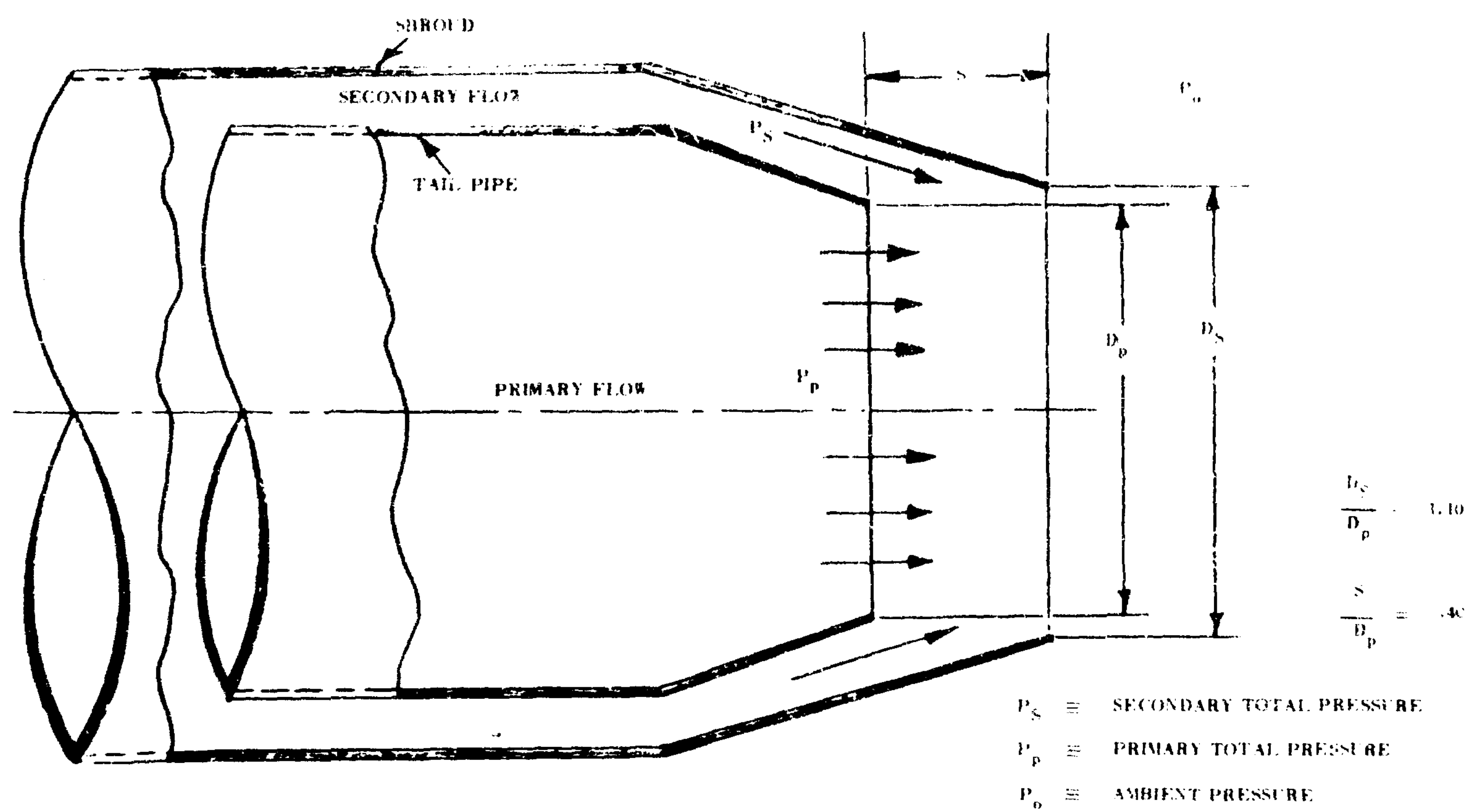


Figure 9. Tail-Pipe Ejector.

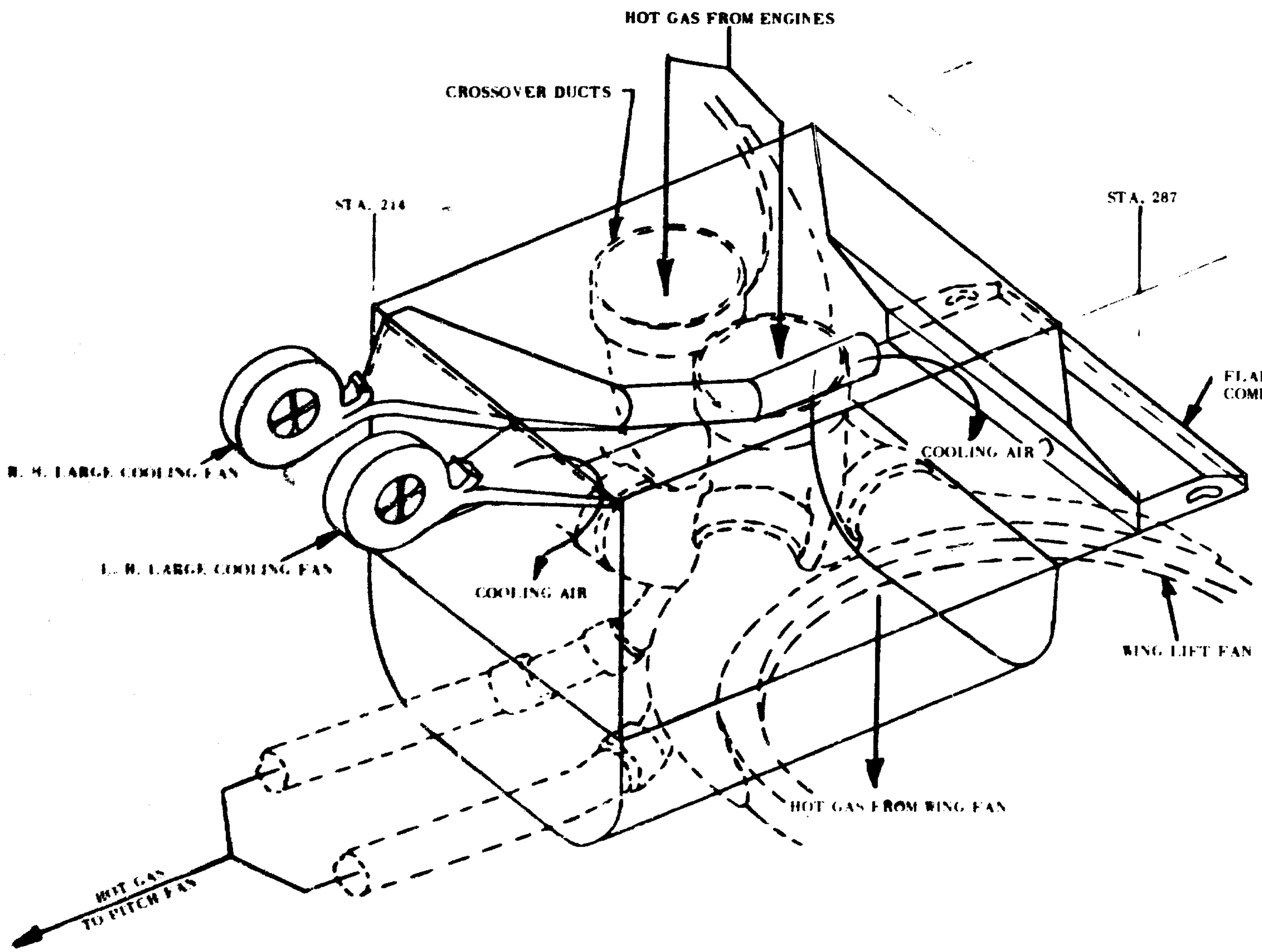


Figure 10. Center Fuselage.

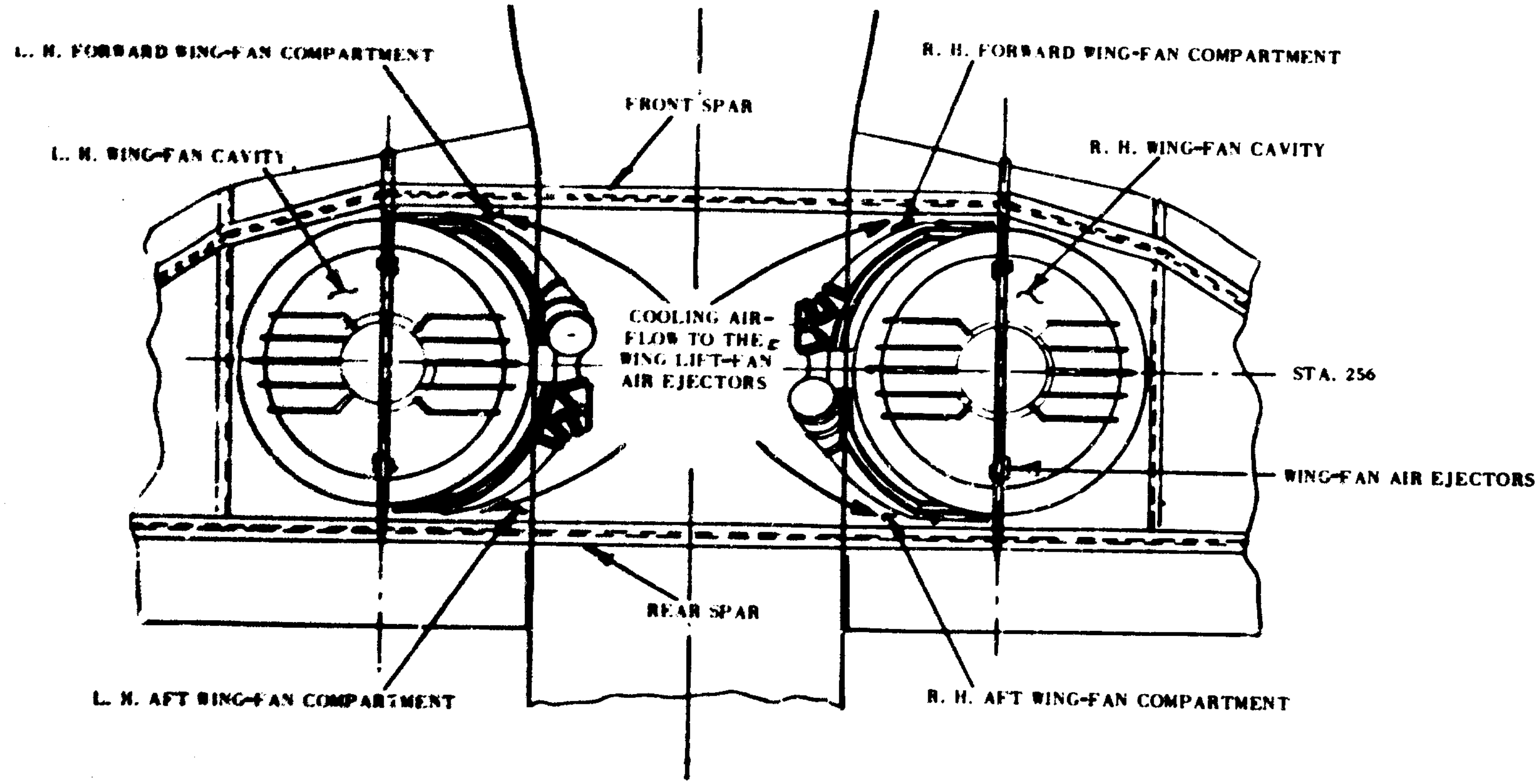


Figure 11. Wing Lift Fans.

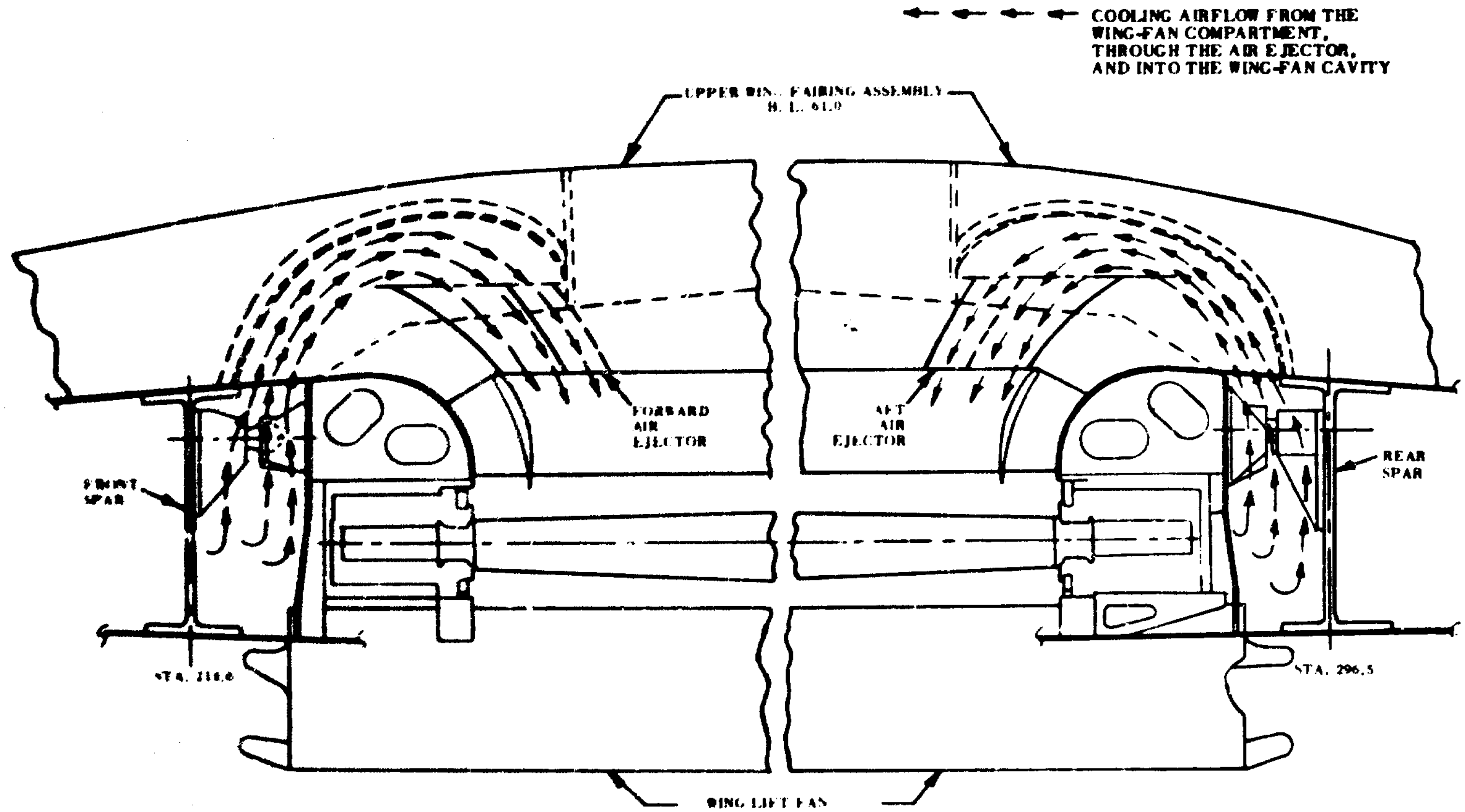


Figure 12. Wing Lift-Fan Air Ejector.

◇ GENERAL ARRANGEMENT

A ENGINE INLET
B BOUNDARY LAYER BLEED DUCT
C COOLING FAN COMPARTMENT AIR INLET
D COCKPIT AIR INLET
E COOLING FANS
F HYDRAULIC PUMP AND GENERATOR
G HYDRAULIC OIL COOLER
H ELECTRONIC BAY
J SMALL COOLING FAN DUCTING
K COOLING FAN COMPARTMENT
L DUCT - COCKPIT TO COOLING FAN COMPARTMENT
M COCKPIT
N AFT PITCH-FAN COMPARTMENT
O NOSE-FAN AIR EJECTOR
P NOSE-FAN CAVITY
Q NOSE FAN
R NOSE-FAN DOORS - LOWER
S NOSE-FAN INLET LOUVERS
T TURBINE CASING
U ENGINE BAY
V DIVERTER VALVE
W TAIL PIPE SHROUD
X TAIL PIPE
Y BOUNDARY LAYER BLEED DUCT FLAPPERS
Z LARGE COOLING FAN DUCTING
AA THRUST SPOILER
BB FLAP ACTUATOR SLOT
CC CROSSOVER DUCTS
DD TAIL PIPE EJECTOR
EE LOWER FUSELAGE CAVITY
FF NOSE-FAN SUPPLY DUCT
GG FORWARD WING SPAR
HH AFT WING SPAR
JJ AFT WING-FAN AIR EJECTOR
KK FORWARD WING-FAN AIR EJECTOR
LL WING FAN
MM WING-FAN SCROLL

○ COOLING AND ENGINE AIRFLOWS

1 OUTSIDE TO COCKPIT
2 OUTSIDE TO BOUNDARY LAYER BLEED DUCT
3 ENGINE INLET
4 OUTSIDE TO COOLING FAN COMPARTMENT
5 SMALL FANS TO HYDRAULIC OIL COOLERS AND ELECTRONIC BAY
6 BOUNDARY LAYER BLEED DUCT TO ENGINE BAYS
7 LARGE COOLING FANS TO ENGINE BAYS
8 SMALL COOLING FANS TO GENERATORS
9 COCKPIT TO COOLING FAN COMPARTMENT
10 LEFT-HAND LARGE BLOWER TO FUSELAGE CAVITY
11 RIGHT-HAND LARGE BLOWER TO FUSELAGE CAVITY
12 ENGINE BAYS TO TAIL PIPE EJECTORS
13 ENGINE DIVERTER VALVES TO CROSSOVER DUCTS
14 CROSSOVER DUCTS TO WING FANS
15 CROSSOVER DUCTS TO NOSE FAN
16 NOSE FAN TO WING FANS
17 NOSE-FAN AFT COMPARTMENT TO NOSE-FAN AIR EJECTORS
18 OUTSIDE TO NOSE-FAN CAVITY
19 NOSE-FAN CAVITY TO OUTSIDE
20 OUTSIDE TO NOSE-FAN CAVITY
21 NOSE-FAN CAVITY TO OUTSIDE
22 FUSELAGE CAVITY TO FLAP ACTUATOR SLOTS TO OUTSIDE
23 ENGINE EXHAUST
24 ELECTRONIC BAY TO AFT NOSE-FAN COMPARTMENT AND FUSELAGE
25 FUSELAGE CAVITY TO FORWARD WING-FAN AIR EJECTORS
26 FUSELAGE CAVITY TO AFT WING-FAN AIR EJECTORS
27 WING-FAN CAVITY TO OUTSIDE

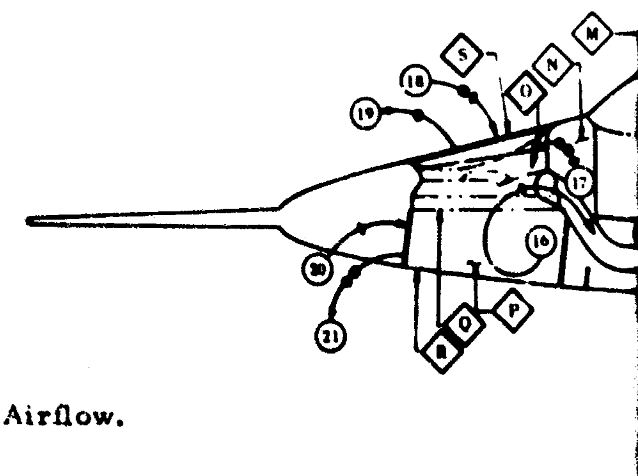


Figure 13. General Arrangement - Cooling and Engine Airflow.

A

LING AND ENGINE AIRFLOWS

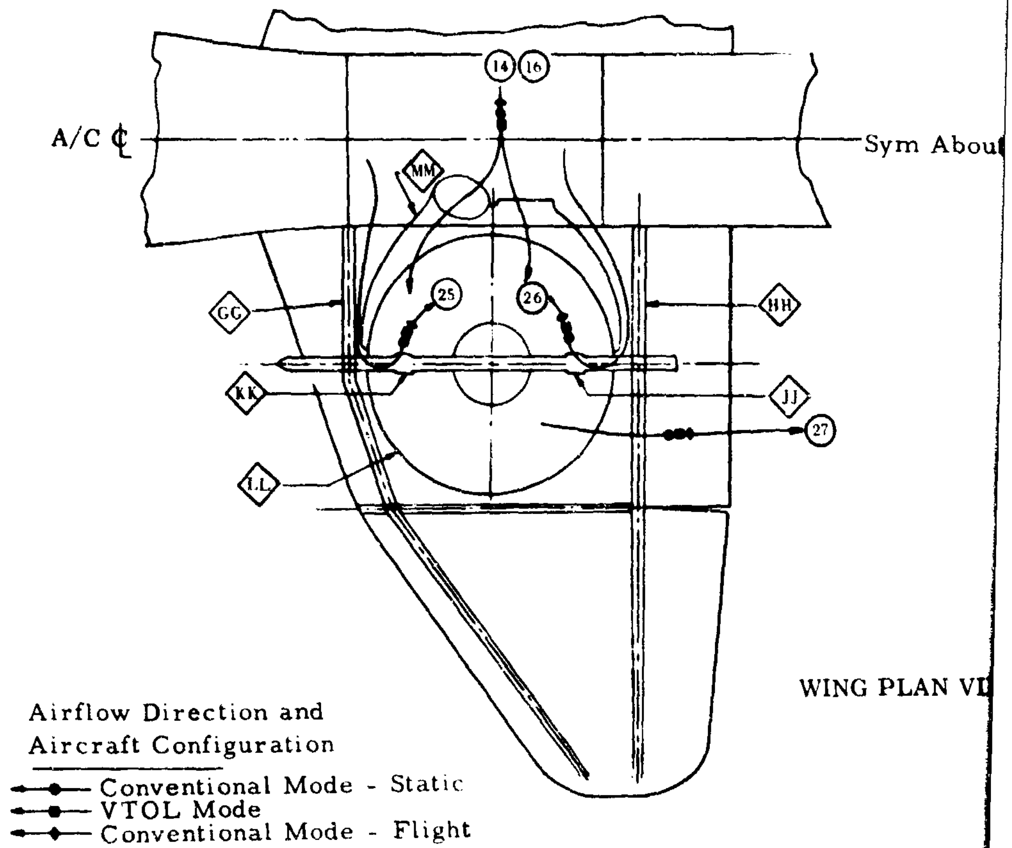
E TO COCKPIT
E TO BOUNDARY LAYER BLEED DUCT
INLET
E TO COOLING FAN COMPARTMENT
FANS TO HYDRAULIC OIL COOLERS AND
RONIC BAY
RY LAYER BLEED DUCT TO ENGINE

COOLING FANS TO ENGINE BAYS
COOLING FANS TO GENERATORS
T TO COOLING FAN COMPARTMENT
AND LARGE BLOWER TO FUSELAGE

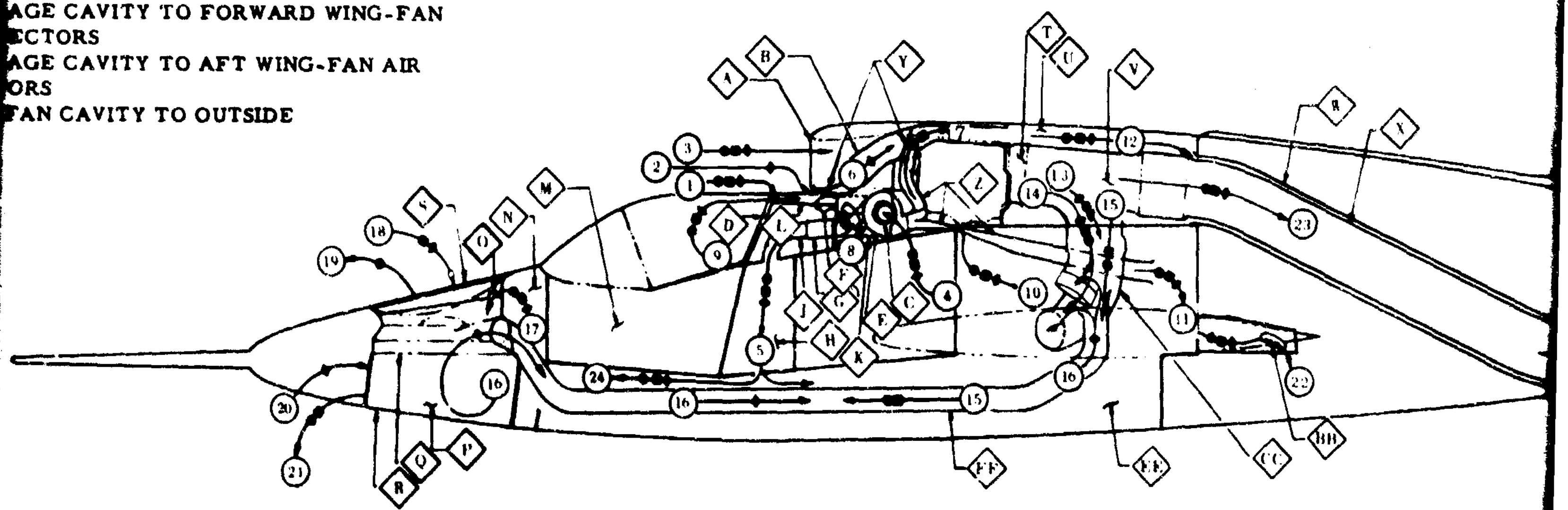
HAND LARGE BLOWER TO FUSELAGE

BAYS TO TAIL PIPE EJECTORS
DIVERTER VALVES TO CROSSOVER

VER DUCTS TO WING FANS
VER DUCTS TO NOSE FAN
AN TO WING FANS
AN AFT COMPARTMENT TO
AN AIR EJECTORS
E TO NOSE-FAN CAVITY
AN CAVITY TO OUTSIDE
E TO NOSE-FAN CAVITY
AN CAVITY TO OUTSIDE
AGE CAVITY TO FLAP ACTUATOR
TO OUTSIDE
EXHAUST
RONIC BAY TO AFT NOSE-FAN
RTMENT AND FUSELAGE
AGE CAVITY TO FORWARD WING-FAN
CTORS
AGE CAVITY TO AFT WING-FAN AIR
ORS
AN CAVITY TO OUTSIDE



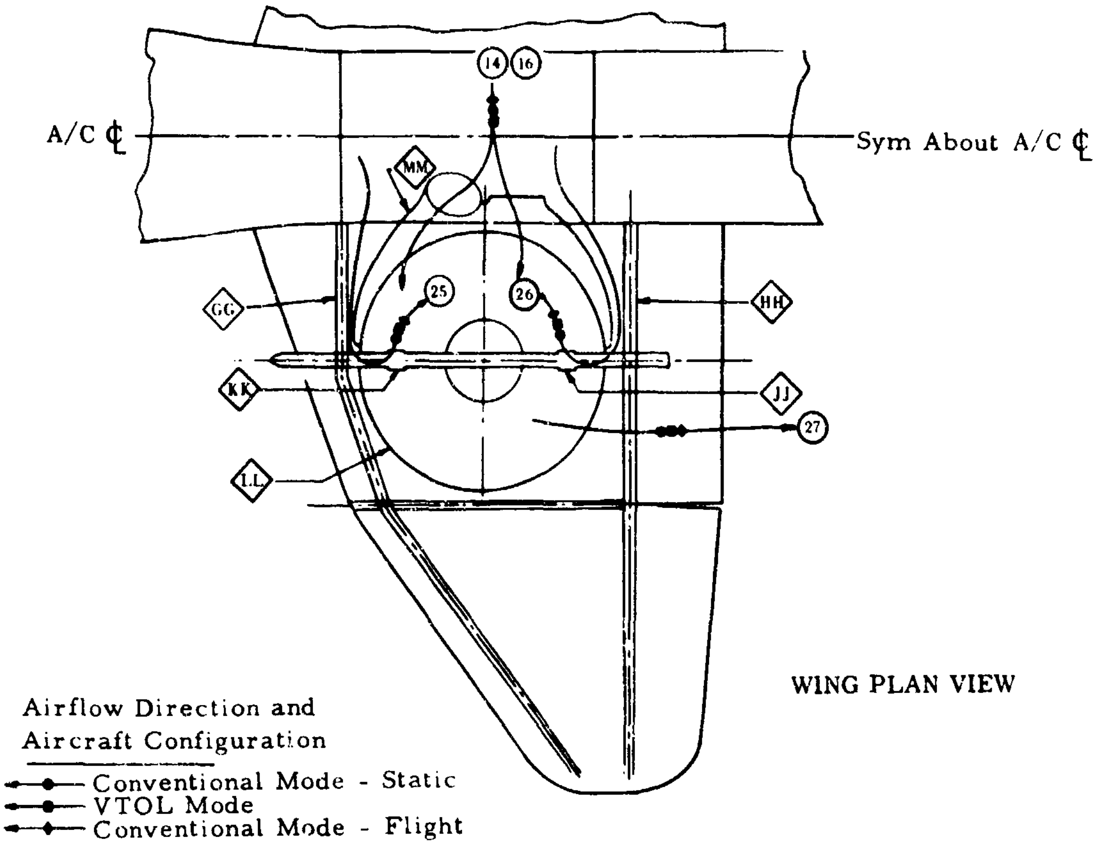
WING PLAN VI



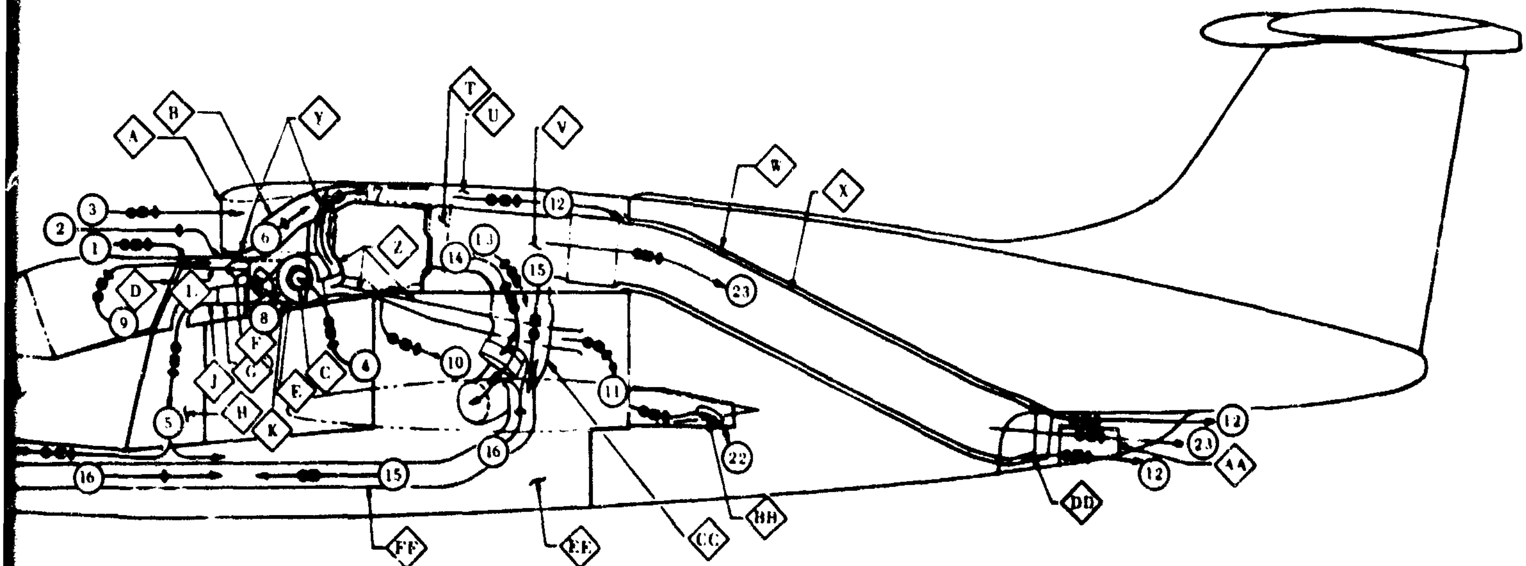
SIDE VIEW

Airflow.

B



WING PLAN VIEW



SIDE VIEW

C

In turbojet-mode flight, two other systems take part in the cooling system. The boundary layer bleed duct opens and supplies cooling air to the engine compartment. As flight speed increases, the boundary layer bleed air overpowers the large blower airflow, closes a flapper valve, and becomes the sole source of engine bay cooling air. With the flapper valve closed, the total large blower output is thus completely diverted to crossover duct cooling and to wing- and nose-fan compartment cooling. In addition, the gas power distribution ducting contributes to cooling system performance during jet-mode operation. This occurs as a result of aerodynamically induced differential pressure between the nose-fan cavity and the wing-fan cavities, which produces a reversed flow through the nose-fan bleed ducts (that is, from nose fan to wing fan). The flow increases with aircraft velocity and is utilized to help counteract diverter valve leakage by cavity purging and direct mixing of cooling air and leakage gases. When diverter valve leakage is within specification limits, sufficient cooling airflow is available for jet-mode (that is, with all fan doors and louvers closed) flight operations above 200 knots. During low-speed flight and during ground operations in the jet mode, the preconversion configuration (that is, with all fan louvers and nose-fan doors open) helps to prevent fan cavity overheating; for example, during high-power/low-speed climbs. Normally, fan cavity overheating during jet-mode operation is corrected by reducing the engine power setting. If further correction is required, it is accomplished by reconfiguring to the preconversion configuration.

Effects of High Temperatures

High-temperature effects on material properties require consideration from two viewpoints: (1) strength level versus temperature and (2) permanent loss of strength due to accumulated soak time. Tables I and II list the design temperature limits and the actual operating temperatures, respectively. While skin materials for the aircraft could withstand the severe thermal environments for the required duration of exposure, the aluminum supporting ribs, frames, and longerons could not withstand the heating without undue loss of strength unless they were protected by some form of insulation or were replaced with materials that were stronger at high temperature. The general location of external construction materials is shown in Figure 14.

Areas subjected to local heating are protected by insulating materials. To provide protection from the intense scrubbing action of hot gases, the insulation blankets are installed with metal edgings and silicone-resin-impregnated skins, or with stainless steel foil, or with elastomeric coatings. Protective coverings are installed in the following locations:

1. Upper closure longeron, nose-fan control door opening.

2. Aft fuselage, thrust spoiler region.

3. Lower wing surfaces, inboard, fore and aft of wing fan and inboard end of flap slot.

4. Fuselage at wing-root lower surface intersection and flap region.

5. Main landing gear struts and braces.

6. Main landing gear doors, forward sections.

Figures 14 through 19 show some of the installation details.

Measurement of Temperature Data

Operational temperature data were measured by use of thermocouples strategically placed throughout the aircraft. The locations and identification of these thermocouples are shown in Figure 20. Temperature measurements were made during ground operation and flight operation. Tables III and IV show the structural and component temperature limits. These data were recorded against real time and were coordinated with other functional and operational data. Direct-reading capabilities were also incorporated to provide monitoring and control of critical temperatures during ground and flight operations. Critical heating conditions refer to those conditions for which there is a definite operating time limit, which, if exceeded, would result in structural, compartment, or component temperatures greater than the established allowable limits. Critical heating conditions can be reached during prolonged hovering in close proximity to the ground and during prolonged high-speed fan-mode flight. The relatively high temperature induced around the aircraft, particularly around the lower surfaces, and the hot gas ingestion by the cooling system produce compartment and component heating during low-level hovering.

Operating Time Limits

Hovering time limit in ground effect is established by electronic compartment inlet air temperature when the aircraft is in the fixed landing gear configuration.

When the aircraft is in the retractable gear configuration with gear extended, hovering time limit in ground effect is established by air temperature in the main landing gear bay. The retractable-gear/gear-extended configuration also establishes the operating time limit for high-speed fan flight due to temperatures in the main gear bay. When the gear is retracted, temperatures at the main landing gear door establish the operating time limit. Similar operating time limits apply to high-speed fan flight in

TABLE I. LOCATION OF DATA ACQUISITION PARAMETERS

Parameter Code*	Limits, °F		Description
	Gnd	Flt	
TG- 4	160	160	Electronics compartment
5	350	300	Upper crossover duct compartment outboard - L. H.
6	↓	↓	Upper crossover duct compartment outboard - R. H.
7	↓	↓	Pitch-fan compartment, cool air inlet - L. H.
8	140	140	Cockpit
11	350	350	Crossover duct compartment
13	-	-	Cooling fan compartment inlet
15	-	-	Cooling fan compartment
16	350	300	Pitch-fan compartment, cool air ejector
17	185	185	Forward cooling fan exhaust - L. H.
19	350	300	Aft cooling fan exhaust - R. H.
21	275	250	Engine compressor section exhaust - L. H.
23	400	400	Engine turbine section - L. H.
24	↓	↓	Engine turbine section - R. H.
25	675	675	Engine exhaust tail pipe ejector temperature - L. H.
27	350	350	Wing-fan aft cooling air ejector - L. H.
28	↓	↓	Wing-fan aft cooling air ejector - R. H.
29	↓	↓	Wing-fan forward cooling air ejector - L. H.
30	↓	↓	Wing-fan forward cooling air ejector - R. H.
42	160	160	Electronics compartment, lower tray, centerline
43	↓	↓	Inverter compartment air
49	325	250	L. H. wing rib, aft upper, 3rd outboard, B. L. 25
812	275	275	MLG wheel-well air temperature - forward
817	↓	↓	MLG wheel well temperature - aft
TL- 1	350	350	Engine lube oil - L. H.
2	↓	↓	Engine gearbox lube oil - R. H.

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43	↓	↓	Inverter compartment air
49	325	250	L. H. wing rib, aft upper, 3rd outboard, B. L. 25
812	275	275	MLG wheel well air temperature - forward
817	↓	↓	MLG wheel well temperature - aft
TL- 1	350	350	Engine lube oil - L. H.
2	↓	↓	Engine gearbox lube oil - R. H.
3	250	250	Hydraulic fluid (reservoir) - L. H.
4	↓	↓	Hydraulic fluid (reservoir) - R. H.
TS-303	700	700	Pitch-fan aft hinge frame - station 85
305	350	350	Pitch-fan bearing temperature
307	300	250	Pitch-fan - front-frame I. D. of casting
310	275	275	Lower forward fuselage longeron - station 82.5
451	700°C	700°C	Engine exhaust duct, station 400 - R. H.
452	325	250	Lower aft fuselage longeron, station 400 - L. H.
455	↓	↓	Aft fuselage, lower longitudinal flange at canted frame 143T005
456	700°C	700°C	Engine exhaust duct, station 400 - L. H.
457	900	700	Aft fuselage vertical stabilizer, front spar frame 143T005, B. L. 6
458	325	250	Aft fuselage canted bulkhead, station 400, B. L. 00
460	300	↓	Diverter valve actuator - L. H.
461	↓	↓	Diverter valve actuator - R. H.
462	650°C	650°C	Engine turbine case flange - L. H.
472	325	250	Aft fuselage canted bulkhead - L. H.
473	↓	↓	Aft fuselage canted bulkhead - R. H.
501	700	400	Space frame 59 member
502	↓	300	SFM 4-20 (space frame)
504	↓	↓	Space frame 79 member
505	↓	↓	Space frame 249 member

*Code: TG - Temperature of gases
TL - Temperature of liquids
TS - Temperature of surface

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B

TABLE I - Continued

Parameter Code	Limits, °F Gnd	Limits, °F Flt	Description
TS-508	325	250	Side skin at MLG door - L. H. *
513	↓	325	Lower longeron skin flange, station 305 (center fuselage section) - L. H.
514	700	300	Space frame - crossover duct area
515	185	185	Fire bottle - L. H.
516	↓	↓	Fire bottle - R. H.
603	325	250	Aft lower cap, B. L. 44 (wing spar) - L. H.
604	↓	↓	Aft lower cap, B. L. 44 (wing spar) - R. H.
605	↓	↓	Aft lower cap, B. L. 61 (wing spar) - L. H.
606	↓	↓	Aft lower cap, B. L. 61 (wing spar) - R. H.
609	↓	↓	Aft upper cap, B. L. 44 (wing spar) - L. H.
610	↓	↓	Aft upper cap, B. L. 44 (wing spar) - R. H.
617	↓	↓	Forward lower cap. B. L. 61 (wing spar) - L. H.
618	↓	↓	Forward lower cap, B. L. 61 (wing spar) - R. H.
619	↓	↓	Wing panel - 5 upper forward inboard - L. H.
622	↓	↓	Wing panel - 6 upper forward inboard - R. H.
623	↓	↓	Wing panel - 7 lower forward inboard - L. H.
624	↓	↓	Wing panel - 8 lower forward inboard - R. H.
625	↓	↓	Wing panel - 9 lower aft inboard - L. H.
626	↓	↓	Wing panel - 10 lower aft inboard - R. H.
630	↓	↓	143W025 bracket 47, lower flange, wing station 268 - L. H.*
644	350	350	Wing-fan bearing temperature - lower - R. H.
651	700	700	Inboard fitting - wing flap - L. H.
701	325	250	MLG support structure - L. H. *
703	↓	↓	MLG drag strut fold joint - L. H. *
705	↓	↓	MLG V-brace - L. H. *
707	↓	↓	MLG mode change cylinder at ring joint*
709	↓	↓	MLG shock strut, wheel - L. H.
711	↓	↓	MLG drag brace at upper pivot - L. H. *

644	350	355	Wing-fan bearing temperature - lower - R. H.
651	700	700	Inboard fitting - wing flap - L. H.
701	325	250	MLG support structure - L. H. *
703			MLG drag strut fold joint - L. H. *
705			MLG V-brace - L. H. *
707			MLG mode change cylinder at ring joint*
709			MLG shock strut, wheel - L. H.
711			MLG drag brace at upper pivot - L. H. *
713		325	MLG door LH-3 inner panel, station 284*
715			MLG door LH-3 inner panel, station 314*
717			MLG door LH-1 inner panel, station 283*
719			MLG door LH-1 inner panel, station 314*
725	275	275	Nose gear wheel well station 106, B. L. 81, W. L. 79*
801	300	300	Landing gear door idler link, forward - L. H.*
802	400	400	Forward landing gear door rod*
803			Landing gear door 031-37 rod*
805	300	300	Landing gear support rod, aft - L. H. *
807	325	325	Aft access panel - L. H. *
811			MLG wheel well heat shield, station 292 - L. H. *
813			MLG wheel well heat shield, station 309 - L. H. *
814			MLG wheel well heat shield, station 333*
911			L. H. compressor case - forward of bleed valve manifold
912			L. H. compressor case - on aft rib of bleed valve manifold
913			L. H. compressor case - aft of bleed valve manifold
914			L. H. compressor case - forward of aft flange
915			R. H. compressor case - forward of bleed valve manifold
916			R. H. compressor case - on aft rib of bleed valve manifold
917			R. H. compressor case - aft of bleed valve manifold
918			R. H. compressor case - forward of aft flange

*Retractable landing gear configuration

TABLE II. OVERTEMPERATURE PARAMETERS

Flight	Parameter Code*	Temperature Limit °F	Temperature Max °F	Duration PCM**Time	Total (sec)	Condition
A90.02G	-	-	-	-	-	CTOL
A90.03G	-	-	-	-	-	CTOL
A90.04G	-	-	-	-	-	CTOL and VTOL
A91F	TS-452	250	305	371-902	530	VTOL maintenance check flight, fixed landing gear
	TS-455	250	274	371-902	5	
	TS-458	250	276	371-902	20	
	TS-610	250	260	371-902	10	
	TS-617	250	276	371-902	50	
	TS-625	250	276	538-751	223	
	TS-514	300	330	527-746	219	
	TG-27	350	360	527-746	10	
	TS-717	325	651	387-949	562	
A92F	TS-307	250	285	1136-1376	240	VTOL level flight performance
	TS-452	250	290	871-1376	505	
	TS-458	250	265	1095-1376	281	
	TS-514	300	320	1058-1376	318	
	TS-625	250	290	944-1376	432	
	TS-790	250	265	1178-1225	47	
	TS-717	325	665	902-1376	474	
A93F	TS-452	250	290	1496-1605	109	CTOL and VTOL maintenance check flight
	TS-455	250	330	736-793	57	
	TS-455	250	280	1269-1285	16	
	TS-455	250	320	1496-1605	109	
	TS-458	250	340	736-793	57	
	TS-458	250	300	1269-1285	16	
	TS-458	250	320	1496-1605	109	
	TS-606	250	300	736-793	57	
	TS-606	250	305	1496-1605	109	
	TS-625	250	275	1532-1605	73	
	TS-711	250	300	736-793	57	
	TS-711	250	270	1269-1285	16	
	TS-711	250	300	1496-1605	109	
	TS-717	325	655	1496-1605	109	
A94F	TS-452	250	299	1633-1949	316	VTOL
	TS-455	250	340	1633-1949	316	
	TS-458	250	346	1633-1949	316	

	TS-717	325	655	1496-1605	109	
A94F	TS-452	250	299	1633-1949	316	VTOL
	TS-455	250	340	1633-1949	316	
	TS-458	250	346	1633-1949	316	
	TS-606	250	312	1675-1949	274	
	TS-625	250	259	1794-1949	155	
	TS-711	250	313	1633-1949	316	
	TS-717	325	630	1654-1949	295	
A95F	TS-307	250	290	1506-1682	176	VTOL
	TS-452	250	290	1308-1688	380	
	TS-455	250	330	1308-1688	380	
	TS-458	250	350	1308-1688	380	
	TS-606	250	310	1308-1688	380	
	TS-625	250	270	1474-1688	214	
	TS-711	250	310	1308-1688	380	
	TS-472	250	350	2353-2421	68	VTOL on ground, L. H. engine onl
	TS-717	325	670	1339-2500	1161	VTOL on ground
A96F	TS-307	250	290	786-1009	312	STOL takeoff and landing performance
	TS-452	250	300	698-1098	400	
	TS-455	250	350	698-1098	400	
	TS-458	250	350	698-1098	400	
	TS-606	250	320	698-1098	400	
	TS-610	250	270	895-1098	203	
	TS-625	250	290	739-1098	359	
	TS-514	300	340	833-1098	265	
	TS-504	300	312	874-1046	172	
	TS-505	300	312	890-1046	156	
	TS-711	250	300	698-1098	400	
	TS-717	325	650	698-1098	400	
A96.05G	-	-	-	-	-	Preconversion (jet mode)
A96.06G	-	-	-	-	-	Preconversion (jet mode)
A96.07G	-	-	-	-	-	Preconversion (jet mode)
A96.08G	-	-	-	-	-	Preconversion (jet mode)
A96.09G	TG-43	160	170	1778-1913	135	VTOL
	TG-43	160	175	2266-2364	98	
	TG-49	250	530	2266-2364	98	
A96.10G	-	-	-	-	-	Preconversion
A98F	TS-303	700	710	2198-2307	111	VTOL
	TS-811	325	350	2333-2343	10	
	TG-49	250	420	2177-2349	172	
A99F	TG-49	250	400	0-963	963	VTOL (hover)
A100F	TG-43	160	180	376-1119	743	VTOL (hover) fixed
	TG-49	250	470	313-828	516	[illegible]

	TS-45[illegible]	250	350	1308-1688	380	
	TS-606	250	310	1308-1688	380	
	TS-625	250	270	1474-1688	214	
	TS-711	250	310	1308-1688	380	
	TS-472	250	350	2353-2421	68	VTOL on ground, L. H. engine onl
	TS-717	325	670	1339-2500	1161	VTOL on ground
A96F	TS-307	250	290	786-1009	312	STOL takeoff and landing performance
	TS-452	250	300	698-1098	400	
	TS-455	250	350	698-1098	400	
	TS-458	250	350	698-1098	400	
	TS-606	250	320	698-1098	400	
	TS-610	250	270	895-1098	203	
	TS-625	250	290	739-1098	359	
	TS-514	300	340	833-1098	265	
	TS-504	300	312	874-1046	172	
	TS-505	300	312	890-1046	156	
	TS-711	250	300	698-1098	400	
	TS-717	325	650	698-1098	400	
A96. 05G	-	-	-	-	-	Preconversion (jet mode)
A96. 06G	-	-	-	-	-	Preconversion (jet mode)
A96. 07G	-	-	-	-	-	Preconversion (jet mode)
A96. 08G	-	-	-	-	-	Preconversion (jet mode)
A96. 09G	TG-43	160	170	1778-1913	135	VTOL
	TG-43	160	175	2266-2364	98	
	TG-49	250	530	2266-2364	98	
A96. 10G	-	-	-	-	-	Preconversion
A98F	TS-303	700	710	2198-2307	111	VTOL
	TS-811	325	350	2333-2343	10	
	TG-49	250	420	2177-2349	172	
A99F	TG-49	250	400	0-963	963	VTOL (hover)
A100F	TG-43	160	180	376-1119	743	VTOL (hover) fixed landing gear
	TG-49	250	470	313-828	515	
	TS-307	250	305	948-1119	171	
	TS-501	400	435	599-838	239	
	TL-1	350	365	563-812	249	
	TL-2	350	353	672-786	214	

*See description of parameters in Table I.
**Pulse Code Modulation.

TABLE II - Continued

Flight	Parameter Code	Temperature Limit °F	Temperature Max° F	Duration PCM Time	Total (sec)	Condition
A101F	TG-43	160	170	776-833	57	CTOL
	TG-5	300	315	1811-2186	375	VTOL fixed landing gear
	TS-504	300	305	2040-2087	47	
	TS-811	325	550	1962-2186	224	
	TL-1	350	375	1650-2186	536	
	TL-2	350	355	1728-2186	458	
	TS-501	400	435	1868-2186	318	
	TS-303	700	720	1676-1967	291	
A102F	-	-	-	-	-	CTOL
A103F	TG-49	250	350	18-65	47	VTOL on ground
	TG-43	160	175	1282-1381	99	VTOL (hover)
	TG-4	160	175	1282-1319	37	
	TG-42	160	215	1319-1381	62	
	TS-801	300	350	1282-1345	63	
	TL-1	350	420	1282-1340	58	
	TL-2	350	400	1319-1350	31	
A104F	TG-4	160	190	1166-1280	114	VTOL
				2560-2732	172	
	TG-49	250	440	2742-3000	268	
	TL-1	350	360	2805-2914	109	
A105F	TG-5	300	310	701-951	250	VTOL (hover) fixed landing gear
	TG-49	250	500	0-1008	1008	
	TL-1	350	370	535-1008	473	
	TL-2	350	360	701-982	281	
	TS-303	700	725	550-738	188	
	TS-501	400	480	535-1008	473	
A106F	TS-501	400	425	2900-3076	176	VTOL fixed landing gear
	TG-49	250	530	1105-1280	175	
				2560-3076	516	
	TL-1	350	360	2900-3056	156	
A107F	TG-49	250	310	1233-1275	42	Maintenance check flight
	TG-49	250	475	2560-3170	610	VTOL (level flight performance)
	TL-1	350	372	2685-3168	483	STOL approach and hover landing
	TS-501	400	407	2903-2976	73	STOL approach and hover landing
A111F	TG-43	160	166	2952-3004	52	VTOL fixed landing gear
	TG-49	250	500	[illegible]	[illegible]	

	TS-501	400	480	535-1008	473	
A106F	TS-501	400	425	2900-3076	176	VTOL fixed landing gear
	TG-49	250	530	1105-1280	175	
				2560-3076	516	
	TL-1	350	360	2900-3056	156	
A107F	TG-49	250	310	1233-1275	42	Maintenance check flight
	TG-49	250	475	2560-3170	610	VTOL (level flight performance)
	TL-1	350	372	2685-3168	483	STOL approach and hover landing
	TS-501	400	407	2903-2976	73	STOL approach and hover landing
A111F	TG-43	160	166	2952-3004	52	VTOL fixed landing gear
	TG-49	250	500	2946-3623	780	
	TG-5	300	310	3290-3490	200	
	TG-6	300	301	3530-3540	10	
	TS-811	325	590	2947-3056	110	
	TL-1	350	360	3358-3623	265	
	TS-501	400	440	3233-3623	390	
A112F	TG-49	250	410	601-913	312	VTOL fixed landing gear
	TG-49	250	280	1188-1344	156	
	TS-709	250	280	601-627	26	
	TG-5	300	330	871-902	31	
	TG-6	300	320	991-1266	275	
	TS-811	325	600	11-91	80	
	TS-811	325	600	601-710	109	
	TS-504	300	320	871-1287	416	
	TL-1	350	354	871-908	37	
	TL-2	350	355	991-1251	260	
	TL-2	350	355	1282-1344	62	
A116F	TG-5	300	330	1929-2043	114	VTOL fixed landing gear
	TG-6	300	410	1825-2043	218	
	TS-504	300	377	1840-2043	203	
	TS-811	325	450	1600-1650	50	
	TL-1	350	370	1710-2043	333	
	TL-2	350	360	1809-2043	234	
A117F	-	-	-	-	-	
A118F	TG-6	300	400	1419-1751	332	VTOL (hover) fixed landing gear
	TS-501	400	470	1320-1751	431	
	TS-811	325	650	1387-1627	240	
	TL-1	350	360	1330-1751	421	
A119F	TS-501	400	480	568-949	381	VTOL fixed landing gear
	TS-504	300	410	490-944	454	
	TS-811	325	700	375-840	465	
	TG-5	300	350	578-918	340	
	TG-6	300	380	469-939	470	

	TG-6	300	301	3530-3540	10
	TS-811	325	590	2947-3056	110
	TL-1	350	360	3358-3623	265
	TS-501	400	440	3233-3623	390
A112F	TG-49	250	410	601-913	312	VTOL fixed landing gear
	TG-49	250	280	1188-1344	156
	TS-709	250	280	601-627	26
	TG-5	300	330	871-902	31
	TG-6	300	320	991-1266	275
	TS-811	325	600	11-91	80
	TS-811	325	600	601-710	109
	TS-504	300	320	871-1287	416
	TL-1	350	354	871-908	37
	TL-2	350	355	991-1251	260
	TL 2	350	355	1282-1344	62
A116F	TG-5	300	330	1929-2043	114	VTOL fixed landing gear
	TG-6	300	410	1825-2043	218
	TS-504	300	377	1840-2043	203
	TS-811	325	450	1600-1650	50
	TL-1	350	370	1710-2043	333
	TL-2	350	360	1809-2043	234
A117F	-	-	-	-	-
A118F	TG-6	300	400	1419-1751	332	VTOL (hover) fixed landing gear
	TS-501	400	470	1320-1751	431
	TS-811	325	650	1387-1627	240
	TL-1	350	360	1330-1751	421
A119F	TS-501	400	480	568-949	381	VTOL fixed landing gear
	TS-504	300	410	490-944	454
	TS-811	325	700	375-840	465
	TG-5	300	350	578-918	340
	TG-6	300	380	469-939	470
	TL-1	350	365	568-949	381
	TL-2	350	351	710-928	218
A120F	TL-1	350	370	740-1290	550	VTOL level flight performance (hover landing)
	TL-2	350	358	854-1290	436
	TG-5	300	350	698-1290	592
	TG-6	300	420	683-1290	607
A121F	TG-43	160	173	561-790	229	VTOL level flight performance (hover landing)

TABLE II - Continued

Flight	Parameter Code	Temperature Limit °F	Temperature Max °F	Duration PCM Time	Total (sec)	Condition
A122F	TG-5	300	313	1186-1290	104	VTOL level flight performance (hover landing)
	TG-6	300	375	786-1311	525	
	TS-501	400	470	853-1327	474	
	TS-504	300	355	822-1327	505	
	TL-1	350	370	869-1327	458	
A124F	TS-310	275	330	952-1135	183	VTOL low altitude translations
	TS-501	400	520	692-1200	508	
	TS-504	300	400	727-1200	473	
	TG-5	300	312	962-1135	173	
	TG-6	300	420	608-1162	554	
A125F	TS-310	275	330	700-882	182	VTOL (hover)
	TG-6	300	420	436-866	430	
	TS-504	300	368	506-887	381	
	TS-501	400	505	506-887	381	
A126F	TS-307	250	279	0-94	94	VTOL fixed landing gear
	TS-310	275	366	411-1066	655	
	TS-504	300	390	546-1071	525	
	TL-1	350	370	551-1071	520	
	TL-2	350	360	770-1071	301	
	TS-501	400	500	562-1071	509	
A127F	TS-310	275	380	454-1089	635	VTOL fixed landing gear
	TG-5	300	338	616-1089	473	
	TG-6	300	456	454-1047	593	
	TS-504	300	400	460-1063	603	
	TL-1	350	380	621-1058	437	
	TL-2	350	362	621-1037	416	
	TS-501	400	498	621-1089	468	
A128F	TG-5	300	315	1586-1706	120	VTOL fixed landing gear
	TG-6	300	385	1498-1706	208	
	TS-504	300	380	1498-1706	208	
	TL-1	350	360	1519-1706	187	
A129F	TS-310	275	370	2061-2410	349	VTOL fixed landing gear
	TS-504	300	400	2212-2420	208	
	TG-5	300	330	2269-2420	151	
	TG-6	300	410	2212-2420	208	
	TL-1	350	370	2067-2420	353	
	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	

	TG-6	300	385	1493-1706	208	
	TS-504	300	380	1498-1706	208	
	TL-1	350	360	1519-1706	187	
A129F	TS-310	275	370	2061-2410	349	VTOL fixed landing gear
	TS-504	300	400	2212-2420	208	
	TG-5	300	330	2269-2420	151	
	TG-6	300	410	2212-2420	208	
	TL-1	350	370	2067-2420	353	
	TL-2	350	355	2212-2420	208	
	TS-717	325	700	2701-2852	151	CTOL (flaps 25 percent)
	TS-457	700	716	2764-2784	20	CTOL (flaps 25 percent)
A130F	TS-310	275	370	1492-1871	379	VTOL fixed landing gear
	TG-5	300	320	1809-1861	52	
	TG-6	300	410	1502-1861	359	
	TS-504	300	370	512-1871	359	
	TL-1	350	360	1414-1861	447	
A131F	TG-5	300	317	1689-1890	120	VTOL fixed landing gear
	TG-6	300	380	1642-1809	167	
	TS-504	300	370	1580-1809	229	
	TS-811	325	440	1356-1809	453	
A132F	TG-6	300	360	983-1177	194	VTOL (hover) landing gear
	TS-504	300	305	1162-1193	31	
	TS-501	400	420	1110-1193	83	
A134F	TS-310	275	310	683-1146	463	VTOL fixed landing gear
	TL-5	300	325	719-907	188	
	TL-6	300	390	641-1141	500	
	TS-504	300	390	673-1120	447	
	TS-811	325	650	777-907	130	
	TL-1	350	360	797-1146	349	
	TS-717	325	600	1250-1307	57	
	TS-501	400	440	777-1146	369	
A135F	TS-310	275	340	597-1310	713	VTOL fixed landing gear
	TS-811	325	580	10-488	478	
	TG-5	300	360	644-1372	728	
	TG-6	300	450	592-1169	577	
	TS-504	300	430	592-1169	577	
	TL-1	350	370	774-1169	395	
	TL-2	350	355	914-1169	255	
	TS-717	325	600	1232-1320	88	
	TS-501	400	480	649-1258	609	
A135.01G	TS-811	325	650	3193-3276	83	Fan mode
A135.02G	TG-43	160	180	1566-3172	06	Fan mode
A135.03G	-	-	-	-	-	Fan mode

	TG-6	300	410	1502-1861	359	
	TS-504	300	370	512-1871	359	
	TL-1	350	360	1414-1861	447	
A131F	TG-5	300	317	1689-1890	120	VTOL fixed landing gear
	TG-6	300	380	1642-1809	167	
	TS-504	300	370	1580-1809	229	
	TS-811	325	440	1356-1809	453	
A132F	TG-6	300	360	983-1177	194	VTOL (hover) landing gear
	TS-504	300	305	1162-1193	31	
	TS-501	400	420	1110-1193	83	
A134F	TS-310	275	310	683-1146	463	VTOL fixed landing gear
	TL-5	300	325	719-907	188	
	TL-6	300	390	641-1141	500	
	TS-504	300	390	673-1120	447	
	TS-811	325	650	777-907	130	
	TL-1	350	360	797-1146	349	
	TS-717	325	600	1250-1307	57	
	TS-501	400	440	777-1146	369	
A135F	TS-310	275	340	597-1310	713	VTOL fixed landing gear
	TS-811	325	580	10-488	478	
	TG-5	300	360	644-1372	728	
	TG-6	300	450	592-1169	577	
	TS-504	300	430	592-1169	577	
	TL-1	350	370	774-1169	395	
	TL-2	350	355	914-1169	255	
	TS-717	325	600	1232-1320	88	
	TS-501	400	480	649-1258	609	
A135.01G	TS-811	325	650	3193-3276	83	Fan mode
A135.02G	TG-43	160	180	1566-3172	06	Fan mode
A135.03G	-	-	-	-	-	Fan mode
A135.04G	TL-6	300	360	1211-1305	94	Fan mode
A135.05G	TS-717	325	729	3623-3731	108	Preconversion
A135.06G	-	-	-	-	-	Preconversion
A136F	-	-	-	-	-	Preconversion

A

TABLE II - Continued

Flight	Parameter Code	Temperature Limit °F	Temperature Max °F	Duration PCM Time	Total (sec)	Condition
A137F	TS-504	300	350	1645-1915	270	VTOL retractable landing gear
	TS-713	325	350	1869-1910	41	
	TS-501	400	410	1640-2004	364	
	TS-717	325	330	1884-1915	31	
A138F	TG-6	300	310	1002-1059	57	VTOL retractable landing gear
	TS-504	300	380	820-1407	587	
	TL-1	350	360	981-1491	510	
	TS-501	400	470	981-1496	515	
A139F	TS-504	300	340	1059-1303	244	VTOL retractable landing gear
	TL-1	350	360	1054-1475	421	
	TS-501	400	450	1028-1475	447	
A140F	TS-504	300	350	807-1462	655	VTOL retractable landing gear
	TL-1	350	370	812-1472	660	
	TS-501	400	460	817-1472	655	
A141F	-	-	-	-	-	VTOL retractable landing gear
A142F	-	-	-	-	-	VTOL gear down
A143F	TG-43	160	173	1878-2959	1081	VTOL & preconversion
	TG-812	275	290	1878-1904	26	VTOL & preconversion gear down
	TG-801	300	330	1430-1441	11	
				1898-1909	11	
	TS-501	400	410	2673-2689	16	
A144F	TG-6	300	310	2065-2143	78	VTOL retractable landing gear
	TS-504	300	330	1977-2159	182	
	TL-1	350	360	1852-2159	307	
A145F	-	-	-	-	-	Conversion techniques
A146F	TS-504	300	420	1192-1519	327	VTOL (hover) gear down
	TS-501	400	470	1192-1519	327	
A147F	TS-504	300	310	2418-2496	78	VTOL (hover) gear up
	TL-1	350	355	2418-2496	78	
A148F	TL-1	350	357	1156-1376	220	VTOL (hover) gear down
	TS-501	400	470	1062-1376	314	

A144F	TG-6	300	310	2065-2143	78	VTOL retractable landing gear
	TS-504	300	330	1977-2159	182	
	TL-1	350	360	1852-2159	307	
A145F	-	-	-	-	-	Conversion techniques
A146F	TS-504	300	420	1192-1519	327	VTOL (hover) gear down
	TS-501	400	470	1192-1519	327	
A147F	TS-504	300	310	2418-2496	78	VTOL (hover) gear up
	TL-1	350	355	2418-2496	78	
A148F	TL-1	350	357	1156-1376	220	VTOL (hover) gear down
	TS-501	400	470	1062-1376	314	
A149F	TS-504	300	319	1157-1256	99	VTOL (hover) gear down
	TL-1	350	356	1126-1261	135	
	TS-501	400	440	1121-1261	140	
A150F	TG-43	160	170	997-1366	369	VTOL gear down
	TS-504	300	315	1262-1366	104	
	TL-1	350	362	997-1366	369	
	TL-2	350	352	1190-1366	176	
	TS-501	400	470	976-1366	390	
	TS-913	250	435	711-1361	650	
A151F	TG-43	160	170	801-1310	509	VTOL gear down
	TL-1	350	365	920-1310	390	
	TL-2	350	356	1066-1310	244	
	TS-501	400	470	931-1310	379	
	TS-913	250	440	671-1310	639	
A152F	TG-812	275	300	610-615	15	VTOL
	TL-1	350	360	953-1322	369	
	TS-501	400	452	932-1333	401	
	TS-913	250	420	615-1322	707	
A153F	TS-913	250	420	617-1607	990	VTOL
A154F	TS-705	250	269	745-766	21	CTOL and VTOL hover takeoff
A155F	-	-	-	-	-	Climb performance, retractable landing gear
A156F	-	-	-	-	-	Climb performance, retractable landing gear

B

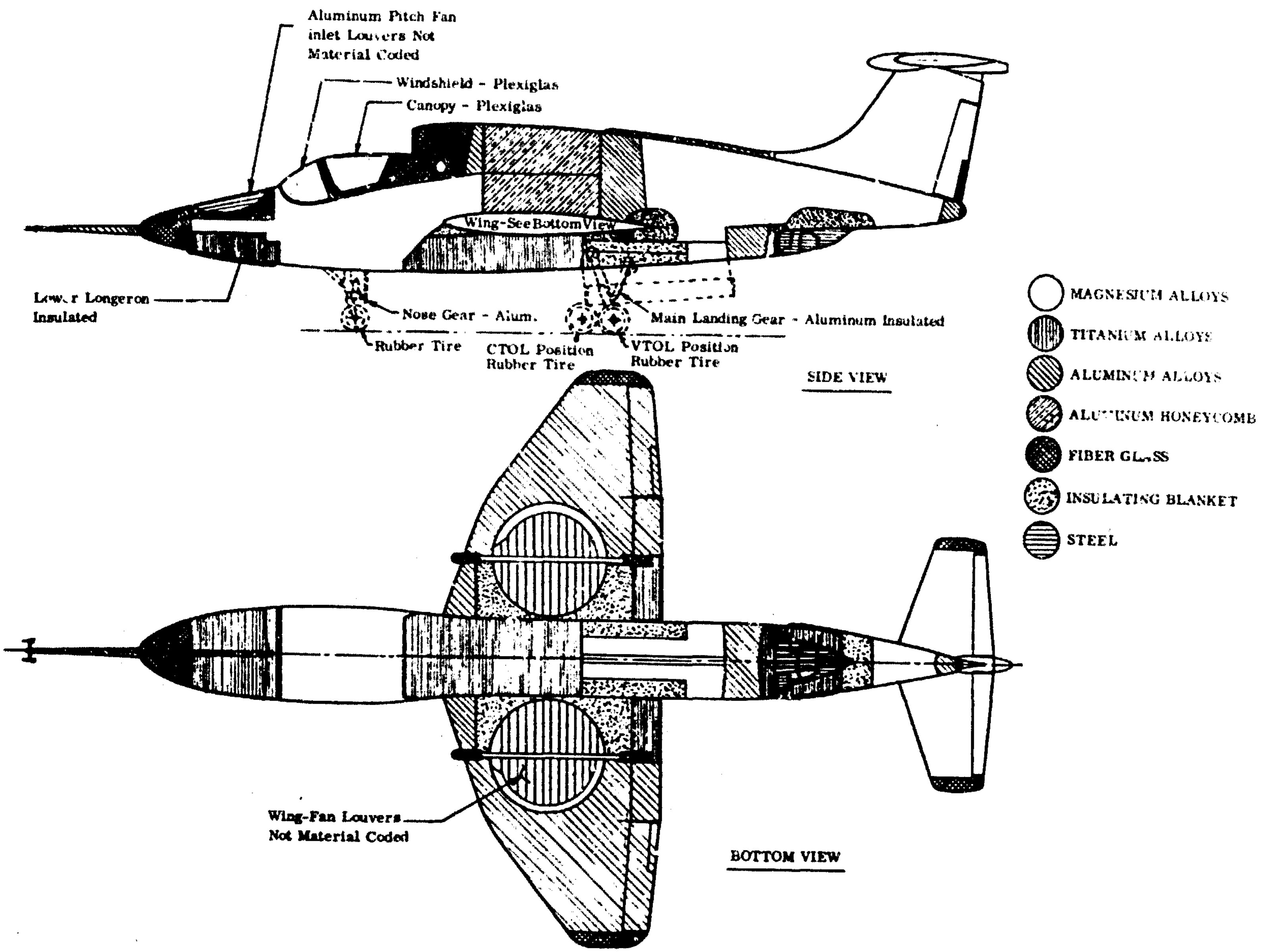


Figure 14. General Location of External Construction Materials.

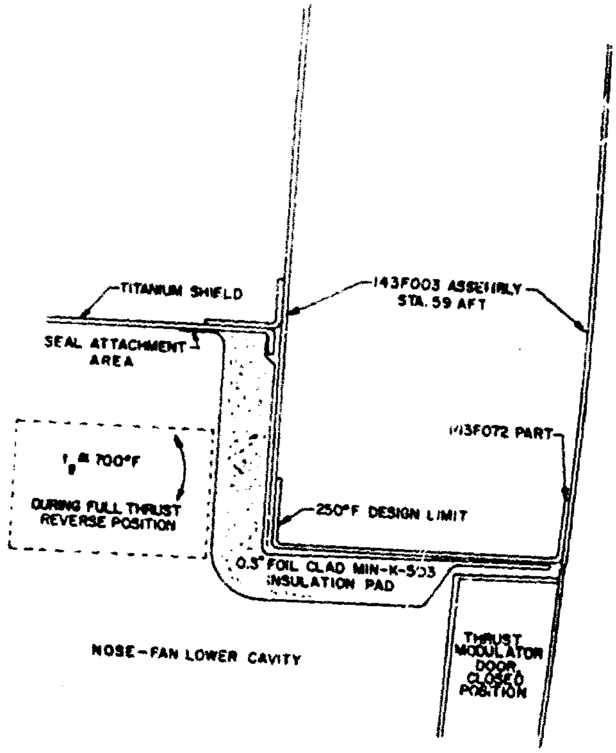


Figure 15. Insulation of Nose-Fan Thrust Reverser Door; Upper Closure Longeron.

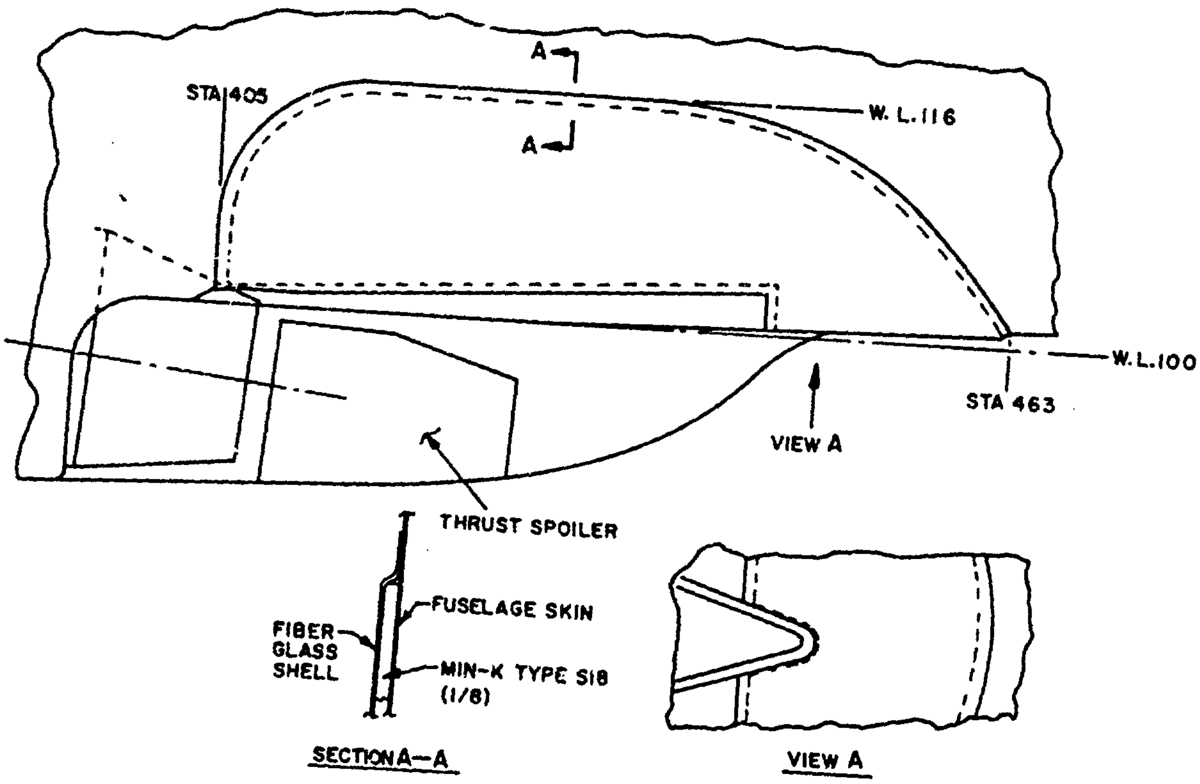


Figure 16. Fuselage Insulation System of Aft Fuselage.

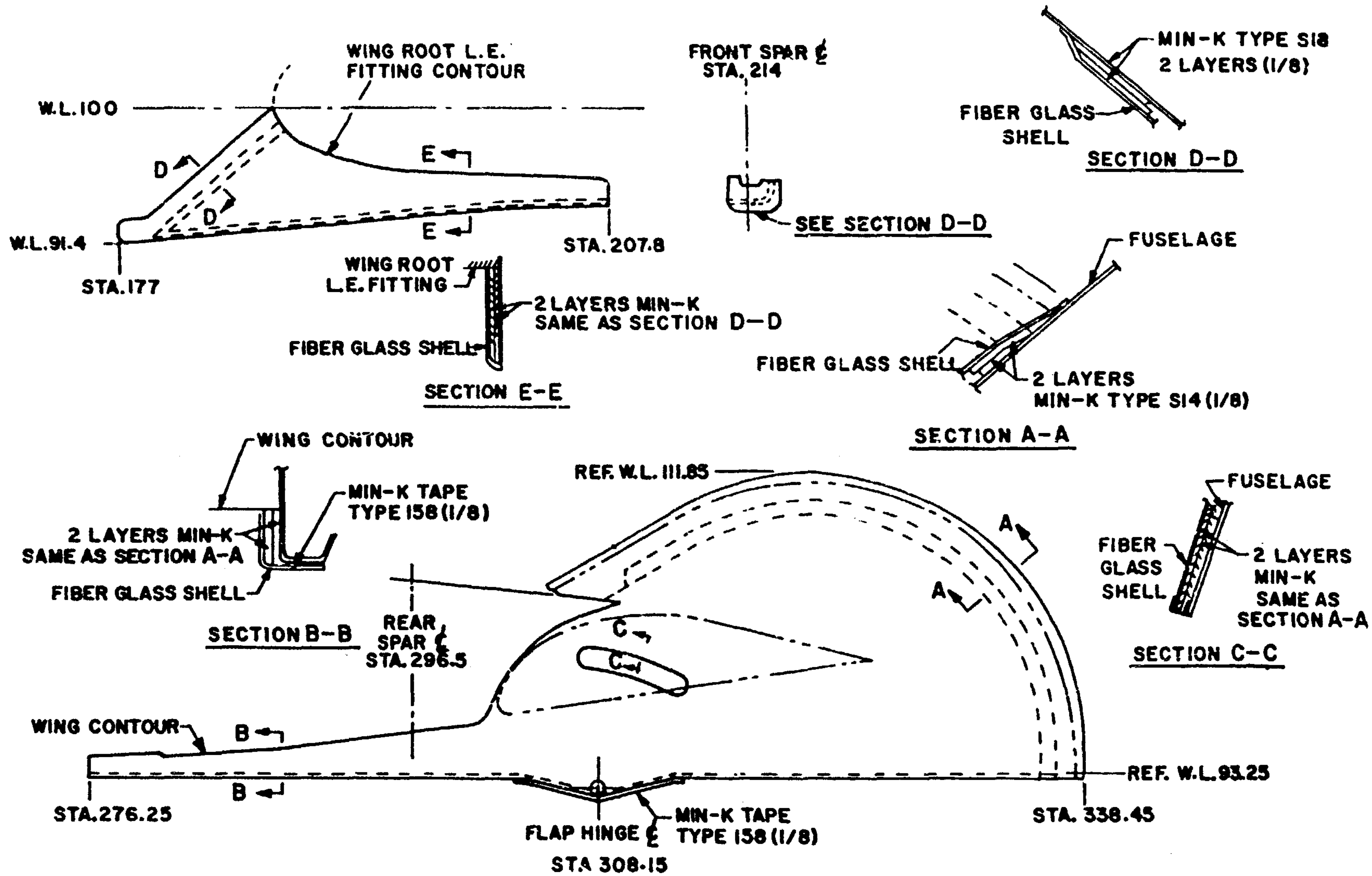


Figure 17. Fuselage Insulation System at Wing Root.

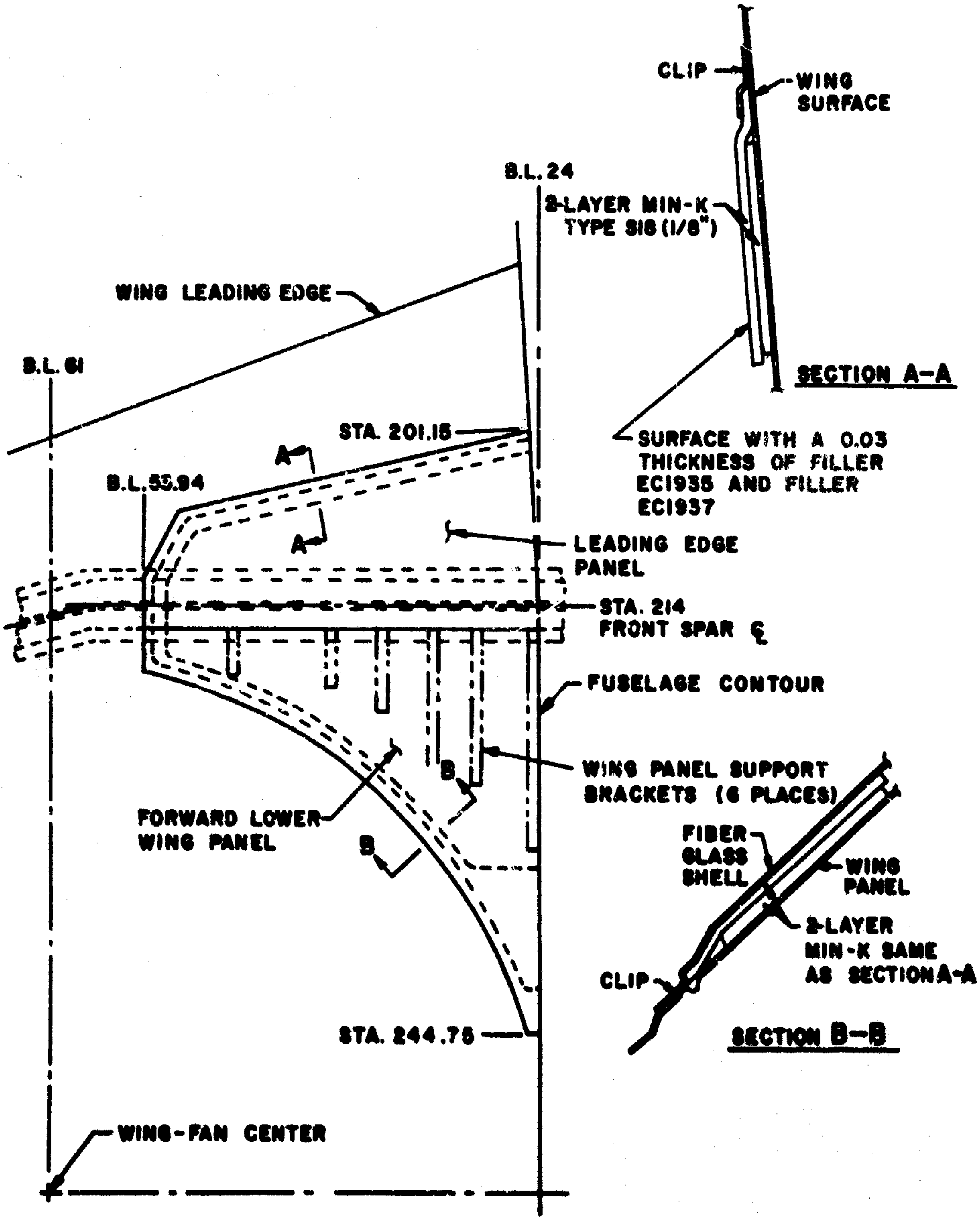


Figure 18. Forward Underwing Surface Insulation System.

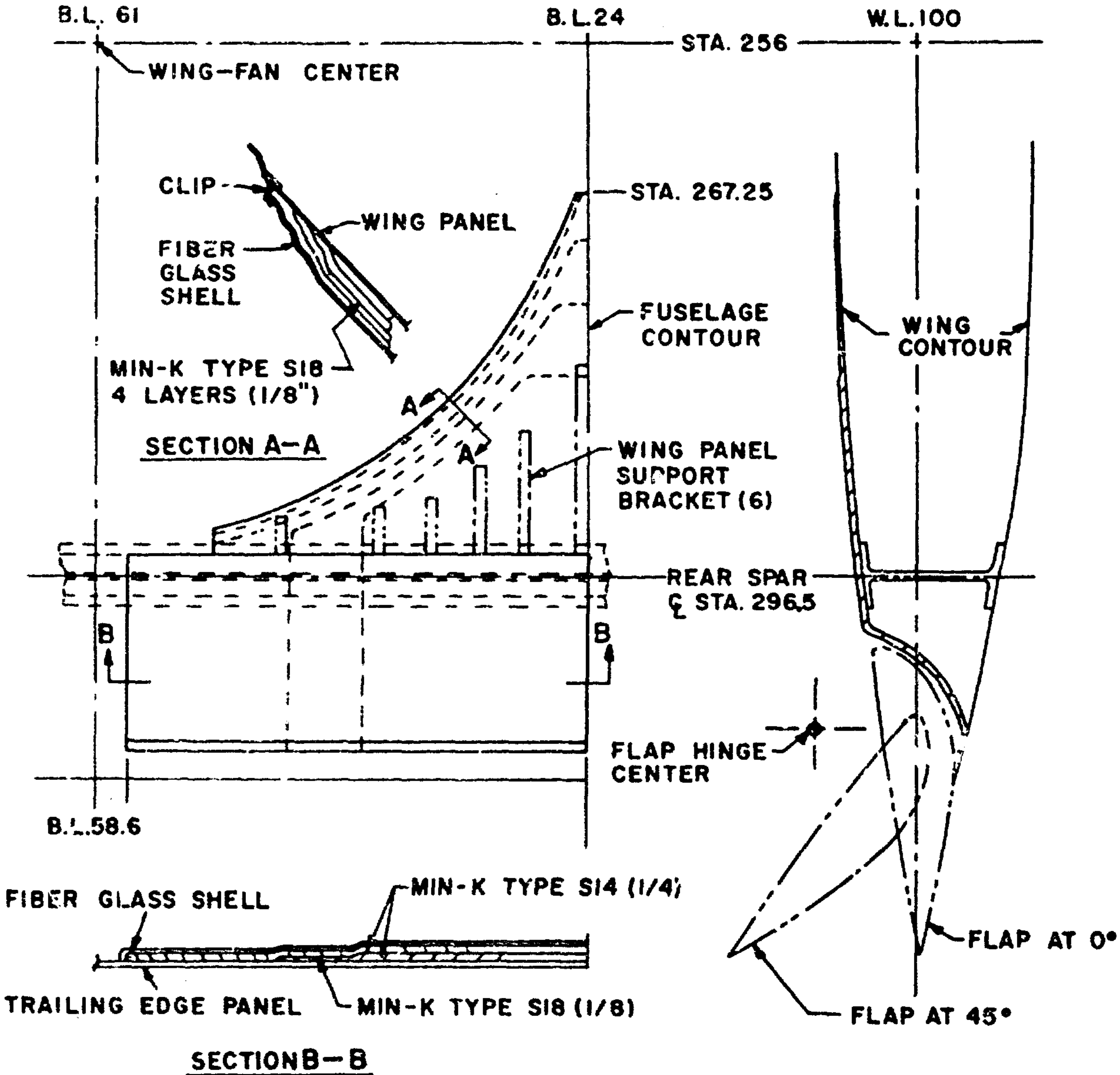


Figure 19. Aft Underwing Surface Insulation System.

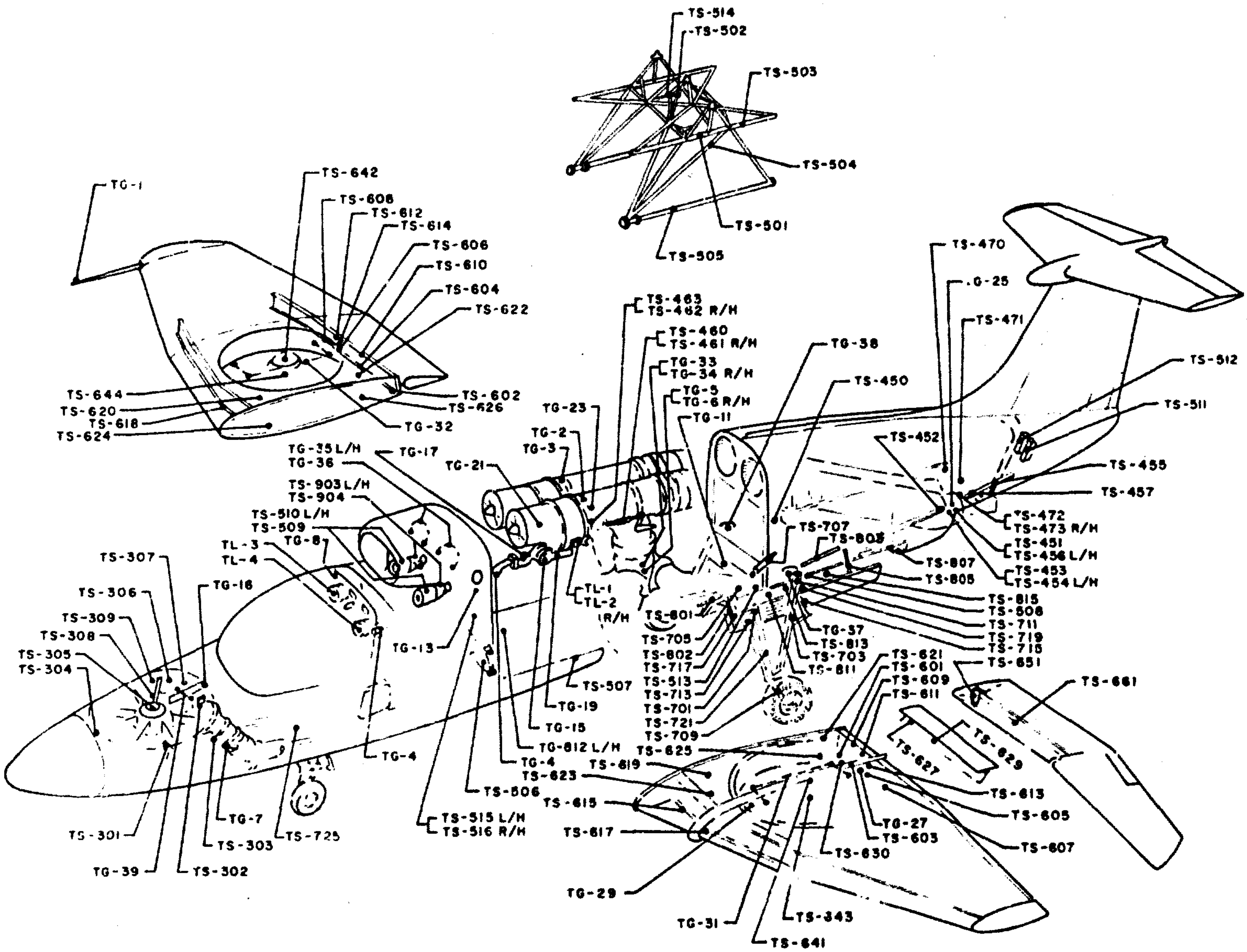
TABLE III. STRUCTURAL TEMPERATURE LIMITS		
Material	Design (Limit) Load (°F)	1g Load (°F)
Aluminum Alloys	250	325
Titanium - 99 T_c	550	1000
6AL4V	700	1100
Magnesium AZ318H24	250	400
Steel-Marage	300	700
Silicone Fiber Glass Laminate	700	700
Silicone Rubber	450	450

the fixed-gear configuration as a result of temperatures in the electronic compartment or wing structure. Wing rib and spar temperatures only occasionally showed a tendency to run high and to limit operating time. This overheating condition may be attributed to hot gas leakage into the wing area through the fan-to-wing finger seals or through the seals at the fuselage-to-canoe junction, to the reduced effectiveness of wing-fan aft cooling air ejectors, or to some combination of these factors.

Fan-mode operating time limits for both fixed and retractable landing gear configurations are shown in Table V. The estimated allowable fan-mode hovering times versus ambient temperature for flight speeds of from 0 to 30 knots are shown in Figures 21 through 24; those for speeds of from 30 to 95 knots are shown in Figure 25.

Cooling Air Inlets

Cooling air inlet locations obviously play a very important role in the performance of the cooling system and are directly affected by aircraft configuration and by those operating conditions that influence the external environment of the aircraft.



THERMOCOUPLE LOCATIONS

FOR WIRE ROUTING SEE 143D020.
FOR WIRING CONN. A/C # 1 SEE 143 D 053 S/2, S/3, S/4.
FOR WIRING CONN. A/C # 2 SEE 143 D 054 S/2, S/3, S/4.
FOR EXACT LOCATIONS REFER TO MARKED PRINTS INDICATED.

Figure 20. Available XV-5A Aircraft Temperature Instrumentation.

SURFACE TEMP. GAUGE IDENT. & LOC. CHART

ITEM NO	MEASURE
TS-601	WING REAR SPAR LNR CAP
TS-602	" " " "
TS-603	" " " "
TS-604	" " " "
TS-605	" " " "
TS-606	" " " "
TS-607	" " " "
TS-608	" " " "
TS-609	" UPPER CAP
TS-610	" " " "
TS-611	" " " "
TS-612	" " " "
TS-613	WING FAN REAR MT. SUPPORT STRUCT.
TS-614	" " " "
TS-615	WING FWD SPAR LWR CAP.
TS-617	" " " "
TS-618	" " " "
TS-619	WING PANEL-3 UPPER FWD. INB'D
TS-620	" -4 " "
TS-621	" -5 UPPER AFT. INB'D
TS-622	" -6 " "
TS-623	" -7 LWR FWD. INB'D
TS-624	" -8 " "
TS-625	" -9 LWR AFT. INB'D
TS-626	" -10 " "
TS-627	WING FLAPFAIRING INB D
TS-629	" " " "
TS-511	INVERTER
TS-512	"
TS-509	GENERATOR
TS-510	"
TS-630	143025-47 BRKT LWR FLANGE
TS-651	FLAP- 143W072 INB'D FITTING
TS-661	" " " "
TS-721	LDG GEAR- TOP OF OLEO STRUT
TS-701	LANDING GEAR - MAIN SUPPORT STRUCTURE
TS-703	" DRAG STRUT FOLD JOINT
TS-705	" BRACE
TS-707	" MODE CHANGE CYL.
TS-709	" AXLE INB'D OF WHEEL
TS-711	" DRAG BRACE UPPER PIVOT
TS-713	LDG GEAR -3 INNER DOOR PANEL-123
TS-715	" " " "
TS-717	" " -1 OUTER PANEL -9
TS-719	" " " "
TS-725	NOSE LANDING GEAR WHEEL WELL
TS-501	SPACE FRAME -59MEM 3-8 UPPER LONG
TS-502	" -41 MEM 4-20 UPPER DIAG.
TS-503	" -73 MEM 8-13 UPPER LONG
TS-504	" -79 MEM 8-13 UPPER LONG
TS-505	" -249 MEM 25-26 LWR LONG
TS-306	PITCHFAN: FRAME OF CASTING
TS-301	" SIDE MOUNT SUPPORT
TS-302	" AFT MOUNT SUPP STRUCTURE
TS-303	" AFT HINGE FRAME
TS-304	" COMPARTMENT FRONT FRAME
TS-305	" BEARING
TS-307	" FRONT FAN FRAME I.D. OF CASTING
TS-308	" FRONT FAN FRAME ON STRUT NEAR HUB
TS-309	" FRONT FAN FRAME ON STRUT NEAR HUB
TS-506	CENTER FUSELAGE LWR WING SPAR CAP
TS-507	LWR ACCESS FAIRING (CANOE)
TS-508	FUS SIDE SKIN - MLG. DOOR SILL
TS-513	CENTER FUS. LWR LONG. SKIN FLANGE
TS-514	SPACE FRAME -46 & 48 -188JCT
TS-641	WING FAN BEARING " " UPPER
TS-642	" " "
TS-643	" " BEARING LOWER
TS-644	" " "
TS-450	AFT FUSELAGE
TS-452	" " LOWER LONGERON
TS-460	DIVERTER VALVE ACTUATOR
TS-461	DIVERTER VALVE ACTUATOR TEMP.
TS-462	LH ENGINE LWR TURBINE SECTION FLANGE
TS-451	EXHAUST DUCT OUTBD SIDE
TS-453	EXHAUST SHROUD " "
TS-454	EXHAUST SHROUD
TS-455	FRAME LWR LNG. AFT FLANGE
TS-457	VERT. STAB FRONT SPAR FRAME
TS-458	AFT FUSELAGE CANTED BULKHEAD

H

GAS & LIQUID TEMP. GUAGE IDENT. & LOC.

ITEM NO.	MEASURE
TL-1	ENGINE OIL (DRAIN TEMP. PLUG)
TL-2	" " "
TL-3	HYD. RES. OIL TEMP.
TL-4	" " "
TG-1	OUTSIDE AIR TEMP.
TG-2	ENG EXHAUST GAS TEMP. (EGT.)
TG-3	" "
TG-4	ELECTRONIC EQUIP. COMP. COOL AIR INLET
TG-5	WING FAN SCROLL COMP. TEMP.
TG-6	" " "
TG-7	PITCH FAN COMP. COOLAIR INLET
TG-8	COCKPIT COMP. AIR TEMP.
TG-11	CROSS DUCT COMP. TEMP.
TG-13	COOLING FAN COMP. AIR INLET
TG-15	" " "
TG-16	PITCH FAN COMP. COOL AIR EJECTOR INLET
TG-17	COOLING FAN (FWD.) EXHAUST TEMP.
TG-19	AFT COOLING FAN EXHAUST TEMP.
TG-21	ENG. COMPRESSOR SECTION
TG-23	ENG. TURBINE SECT. TEMP.
TG-25	ENG. EXHAUST TAILPIPE EJECTOR TEMP.
TG-27	WING FAN COOLING AIR EJECTOR (AFT)
TG-29	" " (FWD.)
TG-31	WING FAN INLET AIR TEMP.
TG-32	WING FAN INLET AIR TEMP.
TG-33	CROSSOVER DUCT INSIDE SHROUD
TG-34	CROSSOVER DUCT INSIDE SHROUD
TG-35	ENGINE INLET AIR TEMP.
TG-36	ENGINE INLET AIR TEMP.
TG-37	FLAP ACTUATOR SLOT IN FUSELAGE
TG-38	FLAP ACTUATOR SLOT IN FUSELAGE
TG-39	
TG-812	MLG. WHEEL WELL AIR TEMP-FWD-C/L
TG-817	MLG. WHEEL WELL AIR TEMP. AFT

ADDITIONAL SURFACE TEMP. GUAGE LOC.

ITEM NO.	MEASURE
TS-801	FWD. LDG. GEAR DOOR IDLER LINK
TS-802	FWD. LDG. GEAR DOOR ROD
TS-803	AFT LDG. GEAR DOOR ROD
TS-805	AFT LDG. GEAR SUPPORT ROD
TS-807	AFT ACCESS PANEL
TS-811	MLG. WHEEL WELL HEAT SHIELD
TS-813	" " " " "
TS-814	" " " " "
TS-456	AFT FUSELAGE EXHAUST DUCT
TS-463	ENGINE LOWER TURBINE SECTION FLANGE
TS-464	ENGINE FUEL CONTROL
TS-465	ENGINE IGNITION BOX
TS-466	ENGINE GEAR BOX
TS-470	AFT FUSELAGE CANTED BLKHD.
TS-471	" "
TS-472	" SKIN
TS-473	" SKIN
TS-515	FIRE BOTTLE
TS-516	FIRE BOTTLE
TS-903	GENERATOR DRIVE GEAR BOX FILLER PLUG
TS-904	" " " "

13

TABLE IV. COMPONENT TEMPERATURE LIMITS

Component	Temperature Limits °C	°F
Gas Generator*		
Exhaust Gas Temperature (EGT)		
Starting 1 second	950	1742
4 seconds	850	1562
11 seconds	750	1382
Steady-State 100 Percent RPM	680	1256
Steady-State Idle	600	1112
Fluctuation	+5, -10	+9, -18
Oil Tank	177	350
Fuel Inlet	43	110
Casing		
Forward Compressor		250
Aft Compressor and Main Frame		750
Combustor		850
Turbine Case		1150
Diverter Valve		1300
Diverter Valve Actuator (hydraulic fluid inlet)		200
Ignition Generator		350
T5 Harness Disconnect		350
Power Pack		300
Tachometer-Generator Alternator		285
Junction Box		300
X353-5B Wing Fan		
Bearing		350
Rotor (turbojet mode)		250
Front Frame (outboard side)		300
X376 Pitch Fan		
Bearings		350

Rotor (turbojet mode)		250
Front Frame (outboard side)		300
X376 Pitch Fan		
Bearings		350
Front Frame		250
Electronic and Electrical Components		
AN/ARC 51X Radio		131
Stability Augmentation System		
Amplifier	80	
Rate Gyro Package	80	
Primary Gain Control Package	80	
Gain Switches		180
Battery - Silver/Zinc (No. 17-S-25)		160
Generator (30 cfm at 131°F) (No. 2CMD99D1)		131
Generator Control Panel (No. 352060DC125A1)		140
(3 ft/sec airflow, s.l. std. day at 140°F or equal)		
(6 ft/sec airflow, s.l. std. day at 160°F or equal)		
Inverter (MS21983-3) (absolute max)	85	121
Circuit Breakers (MP 700)** (MS25244)		180
Wiring		
MIL-W-5086		200
MIL-W-7139		400
MIL-VY-25038		600
MIL-W-16878		400

*Engine components not listed are designed for continuous operation when surrounded by air at an ambient temperature of 250° F.

**This is the preferred part due to separation of power circuit and trip circuit.

B

TABLE IV - Continued

Component	Temperature Limits °C	°F
Electronic and Electrical Components (Continued)		
Switches		
12-HR12-RB (micro)		600
Cockpit (many)		250
Interlock and Limit (many)		400
Actuators		
Wing-Fan Door Locks (SCDE0028-1)		250
Flaps (SCDE0039-1)		250
Thrust Vector (SCDE0045-1)		250
Aileron Droop (SCDE0059-1)		250
Nose-Fan Inlet Louvers (SCDE0066-1)		250
Relays		
Magnetic Latching (BR 9AX-G7-V3)	125	
DPDT (BR7X-300D7-26V)	125	
4PDT (BR14X-1-50B4-26V)	125	
Time Delay (2112-D-H3)		185
Contactor (DH-7L)		250
Connectors		
MIL-C-5015 (high temperature) (T106 type)		800 (cont.) 1200 (S. T.)
MIL-C-5015/MS3190 Pins (CARX type)		257
MIL-C-26482/MS3190 Pins (PTSE type)		257
MS3100K (high temperature) (Cannon) RY 8082-1 and -2, RY 8083-1 and -3		350*
Mechanical and Hydraulic Components		
Ejection Seat (LW-2) (tentative)		160
Cockpit Instrumentation (flight and system indicators)		135
Hydraulic Servo Actuators		

MS3100K (high temperature) (Cannon)	
RY 8082-1 and -2, RY 8083-1 and -3	350*
<u>Mechanical and Hydraulic Components</u>	
Ejection Seat (LW-2) (tentative)	160
Cockpit Instrumentation (flight and system indicators)	135
Hydraulic Servo Actuators	
Exit Louvers (SCDH0002)	275
Nose-Fan Control (SCDH0003)	275
Aileron Boost (SCDH0010)	275
Reservoir (SCDH0005)	275
Pump - Variable Displacement (SCDH0007)	275
Accumulator (MIL-A-8897)	275
Shaft - Accessory Drive (SCDP0021) (tentative)	160
Gearbox - Fan Assembly (SCDP0026)	160
Internal Oil	230
Pump - Fuel Booster (SCDP0029) Inlet Air	550
Valve - Fuel Shutoff	250
Pressure Vessel Fire Extinguisher (SCDP0043/MIL-C-22284)	160
Bearing - Monoball (BAR6445) continuous	800
15 minutes	1000
Bearing - Monoball (2BREM-6A) continuous	800
1 hour	1000

*Continuous operation, 350° F; 5-minute operation, 2000° F; 20-minute operation, no flame passage.

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TABLE V. FAN-MODE OPERATING TIME LIMITS

Fan-Mode Operating Conditions	Retractable Landing Gear			Fixed Landing Gear	
	Gear Position	Time Limit	Temperature-Limiting Factor	Time Limit	Temperature-Limiting Factor
On Ground at 70 Percent J-85 RPM	Down	6.0 minutes	Main landing gear bay	None	-
In Ground Effect at or Near Lift-Off Power	Down	2.0 minutes	Main landing gear bay	7 to 13 minutes* per Figures 21 through 24	Electronic compartment and inlet air
Out of Ground Effect					
0 to 30 Knots	Down	7 to 13 minutes* per Figures 21 through 24	Electronic compartment	7 to 13 minutes* per Figures 21 through 24	Electronic compartment and inlet air
	Up	7 to 13 minutes* per Figures 21 through 24	Inlet air	7 to 13 minutes* per Figures 21 through 24	Electronic compartment and inlet air
30 to 95 Knots (within 1 minute of lift-off)	Down	6.5 minutes* per Figure 25	Electronic compartment or wing structure	6.5 minutes* per Figure 25	Electronic compartment or wing structure
	Up	6.5 minutes* per Figure 25	Electronic compartment or wing structure	6.5 minutes* per Figure 25	Electronic compartment or wing structure
Above 60 Knots	Down	15 seconds, maximum	Main landing gear bay	6.5 minutes* per Figure 25	Electronic compartment or wing structure
	Up	4.0 minutes	Main landing gear doors	6.5 minutes* per Figure 25	Electronic compartment or wing structure

*General conditions at Edwards Air Force Base permitted an operating time limit of 7 to 13 minutes.

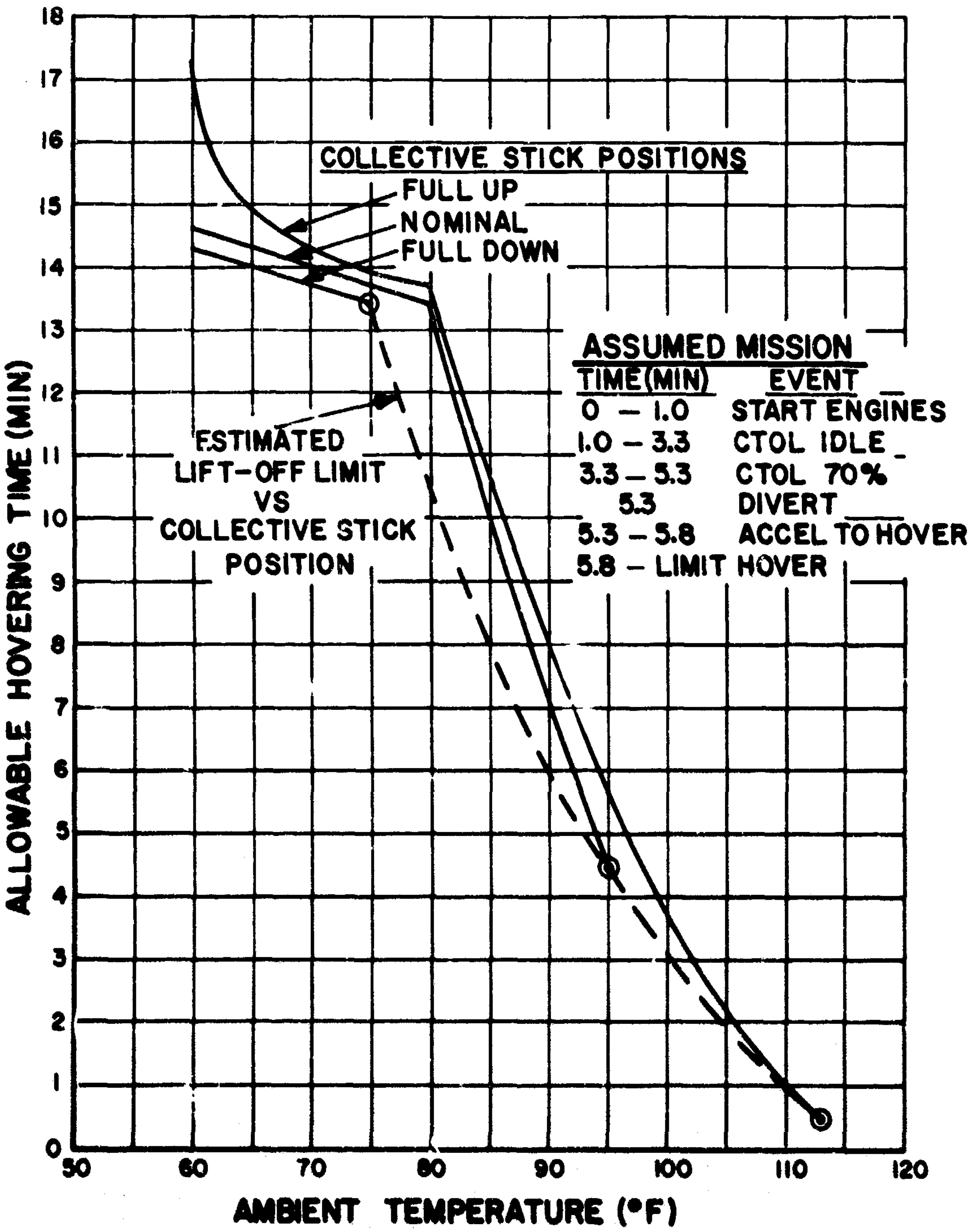


Figure 21. Estimated Allowable Fan-Mode Hover Time When h/D = 2.0, Aircraft Weight Is 10,000 Pounds, Speed Is From 0 to 30 Knots, and Altitude Is 2500 Feet.

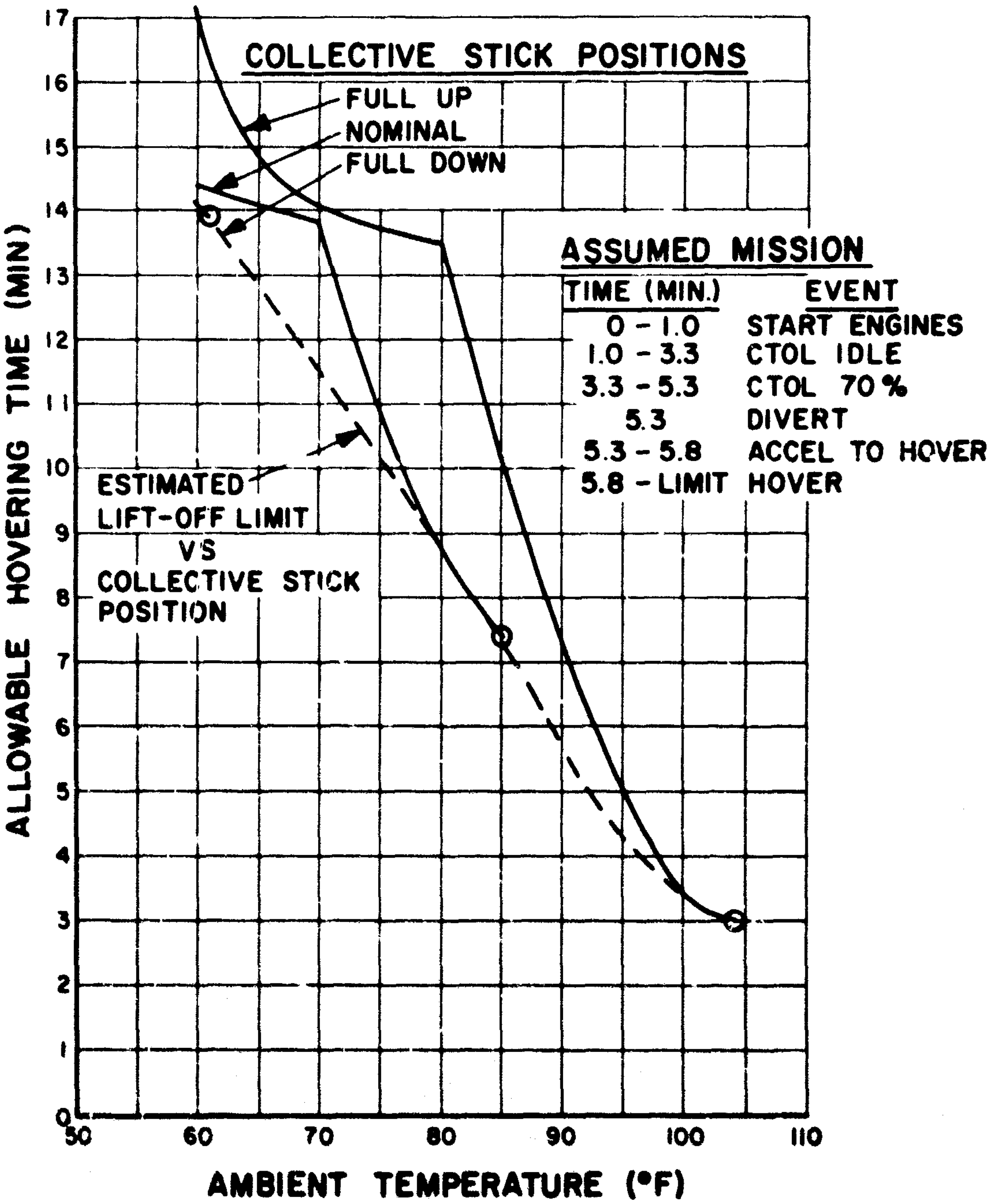


Figure 22. Estimated Allowable Fan-Mode Hover Time When h/D = 2.0, Aircraft Weight Is 10,500 Pounds, Speed Is From 0 to 30 Knots, and Altitude Is 2500 Feet.

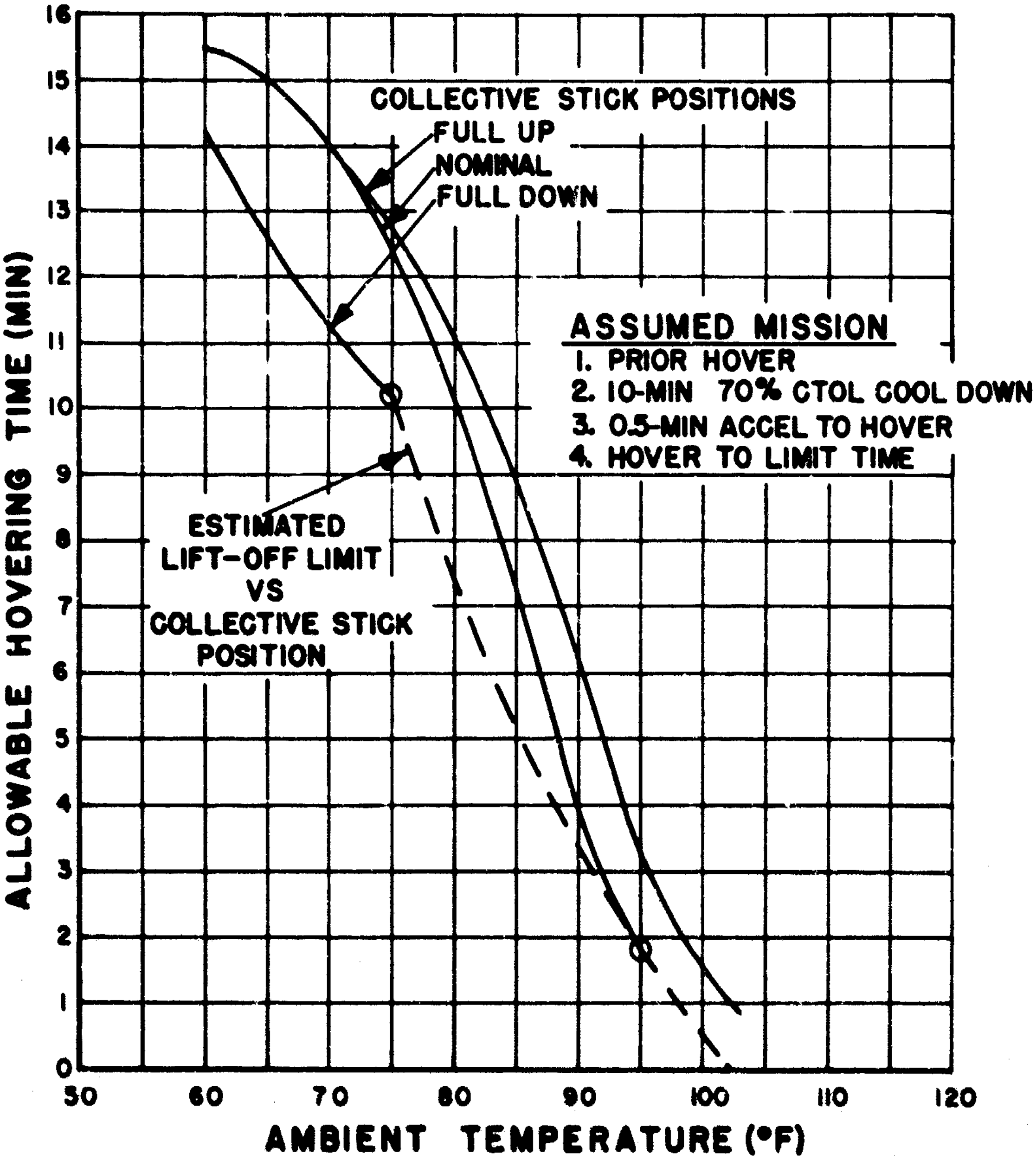


Figure 23. Estimated Allowable Fan-Mode Hover Time When h/D = 1.0, Aircraft Weight Is 10,000 Pounds, Speed Is From 0 to 30 Knots, and Altitude Is 2500 Feet.

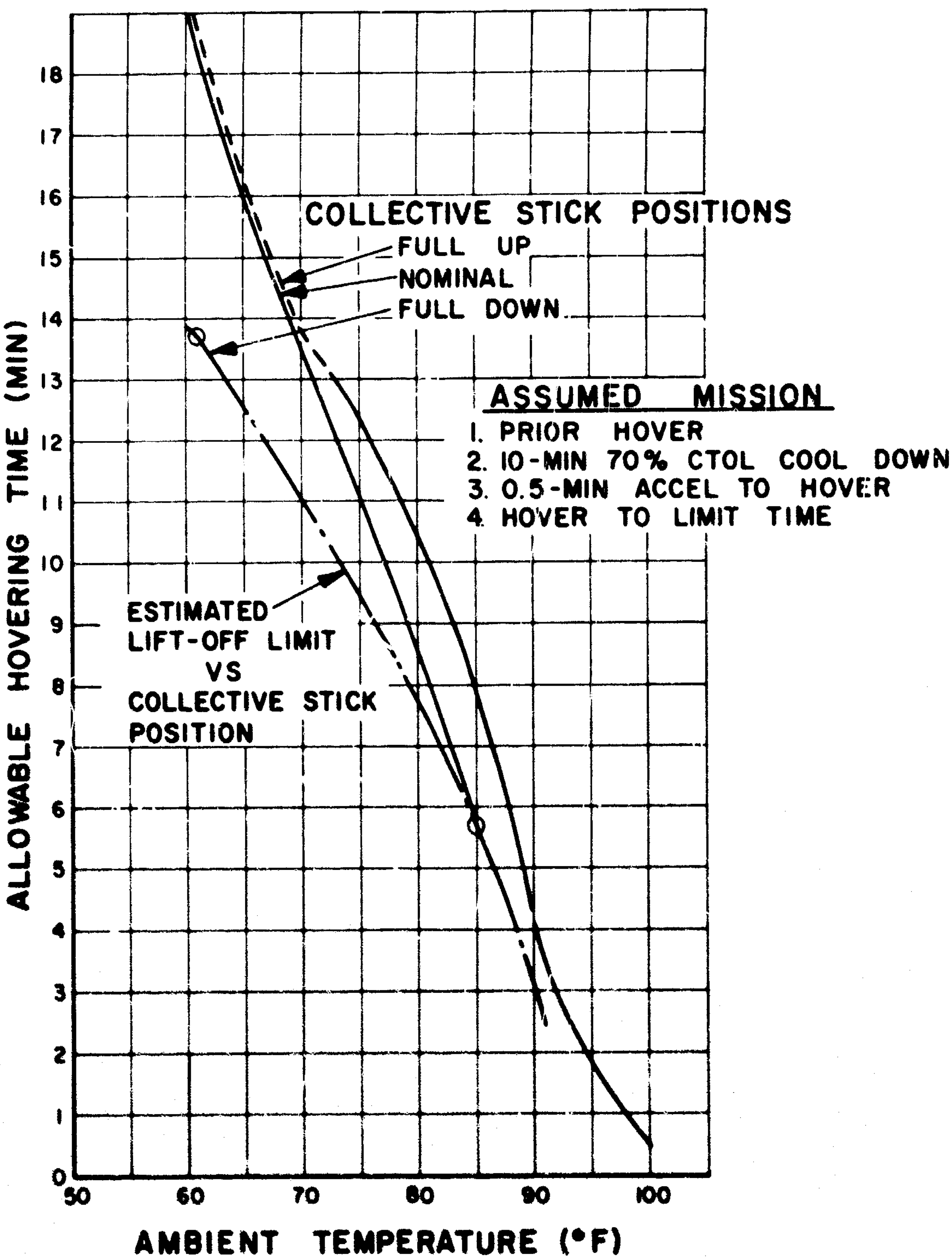


Figure 24. Estimated Allowable Fan-Mode Hover Time When h/D = 1.0, Aircraft Weight Is 10,500 Pounds, Speed Is From 0 to 30 Knots, and Altitude Is 2500 Feet.

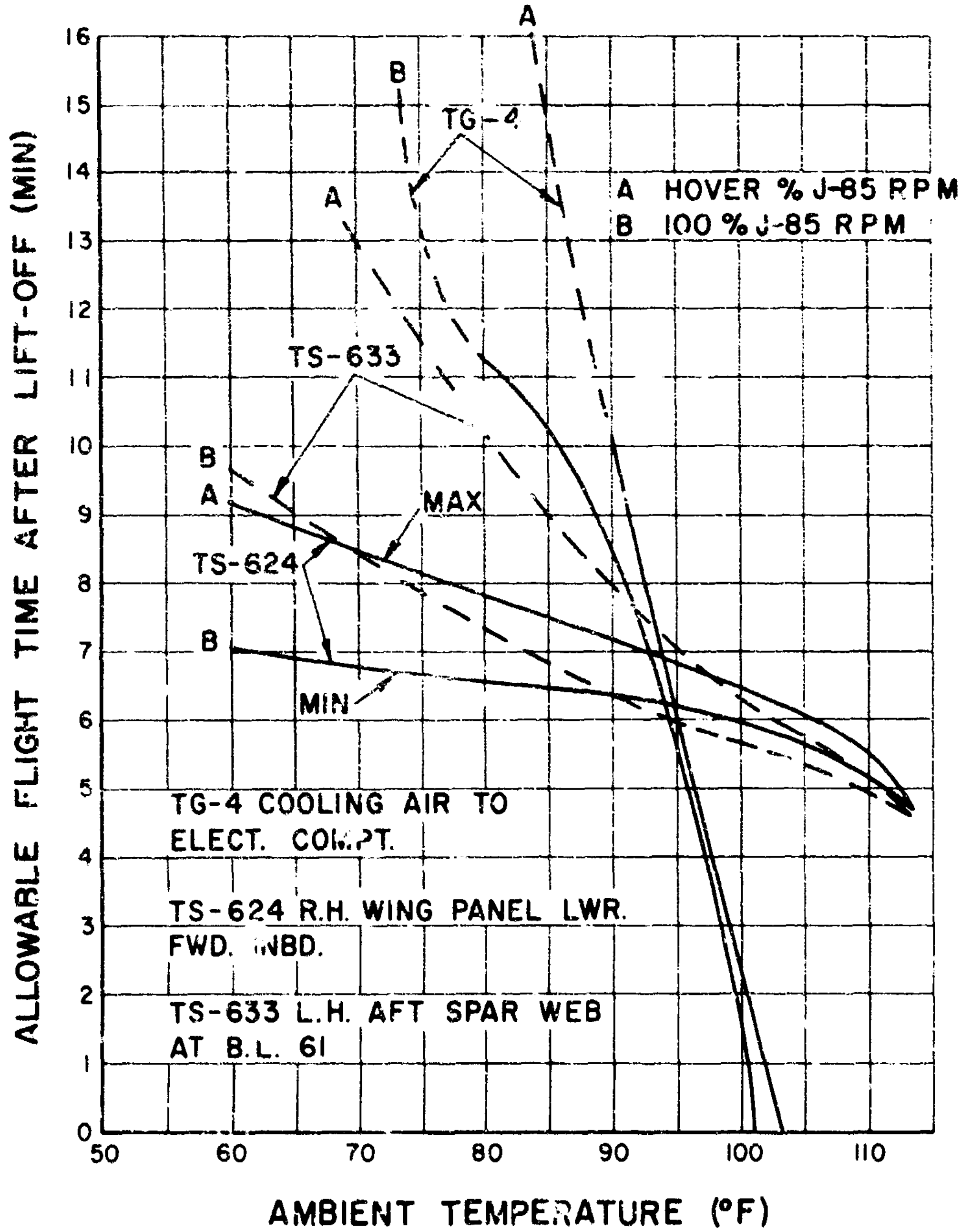


Figure 25. Estimated Allowable Fan-Mode Flight Time When Speed Is From 30 to 95 Knots and Altitude Is 2500 Feet.

Temperature in Cockpit

Cockpit temperature control was not provided. Pilots reported discomfort due to both excessively high and excessively low cockpit temperatures. Maintaining cockpit temperatures within the comfort zone would have a beneficial effect.

Temperature in Electronic Compartment

Except for the environment of the cockpit, that of the electronic compartment is most sensitive to hot gas ingestion. This air is a blend of the air from the cockpit and the cooling fan compartment and of the generator cooling air that has passed through the small cooling air blower and the hydraulic oil cooler. As indicated in Table V, electronic compartment temperature is one of the limiting factors of allowable fan-mode operating time.

A more important aspect of the allowable electronic compartment temperature (160° F) is that the AN/ARC 51X communication radio installed there should not be subjected to an ambient temperature greater than 131°F. This overtemperature condition adversely affects radio reliability.

Temperature in Aft Equipment Compartment

The aft equipment compartment is not a part of the forced-air cooling system. Due to high residual thrust at idle power, the thrust spoilers were used to reduce taxi braking effort. Extended taxiing required at Edwards Air Force Base and some prolonged ground test runs have resulted in compartment temperatures approaching, and occasionally exceeding, the allowable limit. This has had a deleterious effect on the inverters located in that compartment.

Temperature in Wing-Fan Cavity

Wing-fan cavity overheating normally is encountered during high-power/low-speed climbs (100 to 200 knots). It has also occurred during high-speed level flight (300-plus knots). The removal of two exit louver tips from each wing fan to promote hot gas purging helped to reduce the overheating condition.

MAINTAINABILITY HISTORY AND RECOMMENDATIONS

Man-Hours Expended on Maintenance

The man-hours expended in correcting discrepancies in the airframe cooling system are shown in Table VI.

TABLE VI. MAN-HOURS EXPENDED IN RESTORING AIRCRAFT

Part Name	Preparation	Diagnosis	Accessibility	Corrective Action	Reassembly	Check-out	Total Man-Hours	Remarks
Bulkhead at Frame 71	0.1	0.1	0	2.0	0	0	2.2	Resealed
Right End Assy, Hydraulic Compartment Bulkhead	0.1	0.1	0.5	0.3	0.7	0	1.7	Stop-drilled crack
Former at Station 201.9	0.1	0.2	0.2	0.6	0.2	0	1.3	Cleared interference point
Angle, Fire Wall in Engine Compartment	0.2	0.1	0.1	2.3	0.1	0.1	2.9	Replaced because of wear

Efficiency of Insulation

External structural insulation has provided thermal protection for the aircraft structure during this flight test program. Though no quantitative analysis has been conducted, it is likely that some performance degradation has resulted from this insulation. The effects may have been small. However, the nuisance effects on maintenance requirements are more pronounced. Accessibility for inspection and maintenance is very difficult in some areas. Problems range from special care required for disassembly and reassembly during frequent minor repairs to destructive disassembly for inspection and maintenance.

Each insulation detail element has its own peculiar set of performance requirements and operating conditions that are predicated on aircraft configuration and mission requirements. A trade-off study to evaluate the effects of design changes on aircraft performance, mission profiles, maintenance requirements, and reliability is recommended. These changes would include elimination of insulation, substitution of materials capable of withstanding higher temperatures, and/or the redesign of detail elements. The study should also include an evaluation of the trade-offs between weight and maintainability. The objective of this study would be to establish criteria for potential derivative aircraft.

Temperature Control

The average maximum temperatures of all parameters recorded under all flight conditions varied from 170° F in the electrical inverter compartment to 729° F at the inner panel of the main landing gear door; the average of all maximum temperatures recorded was 365° F. The time during which maximum temperatures existed was from 5 seconds to 1,170 seconds

(19.5 minutes), the average of the time periods during which high temperatures exceeded established limits was 4.85 minutes.

Cooling Air Inlets

The cooling blower plenum inlets on the XV-5A are subject to hot gas ingestion, which degrades cooling system performance. It is recommended that a study be conducted to determine if a more favorable location for these inlets is possible and/or if other methods of eliminating or alleviating hot gas ingestion are feasible.

Cockpit Heating

Cockpit heating has been aggravated by hot gas ingestion through the cockpit fresh-air inlet (see Figure 7), particularly during prolonged operation in ground effect. Relocation of the fresh-air inlet to a position free of (or at least less subject to) hot gas ingestion and incorporation of cockpit insulation and pilot-operated ventilation controls would improve the cockpit environment.

Aft Equipment Compartment

Cooling system versus inverter modification trade-offs versus operating requirements should be reviewed to determine appropriate corrective action to improve inverter reliability.

DESIRABLE FEATURES

1. The ejectors for the tail pipe, wing fan, and nose fan provide an effective, lightweight cooling system airflow augmentation.

2. The boundary layer bleed system was required to achieve engine inlet performance. Effective use of this bleed air has been made for the cooling system airflow augmentation.

UNDESIRABLE FEATURES

1. Overheating in wing-fan cavity.

2. No temperature control in cockpit.

3. Unsatisfactory seals against hot gases (fan-to-wing finger seals; tail-pipe ejector nozzle-to-fuselage seals; and seals at the fuselage-to-canoe junction at the hinge lines and edges of the

main landing gear door, at the diverter valve door, and at the nose- and wing-fan scroll).

4. Excessive temperatures in electronic compartment and AN/ARC 51X radio during fan-mode operation for both ground and flight test operations.

5. Ingestion of hot exhaust gases during operation in the fan mode because of the location of cooling system plenum inlets.

6. Insufficient cooling of aft equipment compartment and components.

7. Maintenance difficulties caused by external insulated structures.

RECOMMENDATIONS AND SUGGESTIONS FOR UNEVALUATED IMPROVEMENTS

Considerations in Determining Cooling Requirements

In addition to providing for normal operations, the basic design adopted for the cooling system must also provide additional temporary protection, as required, to accomplish ground and flight operations safely. The following parameters should be considered, since they affect cooling requirements:

1. Mode of aircraft operation (that is, fan mode or turbojet mode).

2. Aircraft speed.

3. Power setting.

4. Positions and movement of controls.

5. Aircraft in or out of ground effect.

6. External environment.

7. Time duration of given set of conditions.

Operating Limits for Maintenance of Temperatures

The general requirement of the cooling system and of the structural insulation is to maintain structural and component temperatures within safe operating limits during all phases of intended aircraft operation. To determine the safe operating limits, the following operating conditions, which

are listed in increasing order of severity, must be considered:

1. Normal operation.

2. Prolonged fan-mode flight out of ground effect with wing-fan louvers vectored at less than 15^{o}.

3. Prolonged fan-mode flight out of ground effect with wing-fan louvers vectored at greater than 15^{o}.

4. Prolonged fan-mode operation in ground effect.

Structural High-Temperature Limits

Fan-mode operation is characterized by more severe heating conditions than jet-mode operation and also by lower maximum possible maneuvering loading conditions. Two sets of allowable structural high-temperature limits, related to the two modes of aircraft operation, were established as operational limits: (1) the design- (or limit-) load conditions and (2) the 1g-load conditions. Design-load and 1g-load temperature limits for structural materials are shown in Table III. A number of aircraft component temperature limits are shown in Table IV. The general location of external aircraft construction materials is shown in Figure 14.

AIRFRAME STRUCTURAL OVERHEAT WARNING SYSTEM

SYSTEM CONFIGURATION AND OPERATION

The structural overheat warning system (see Figure 26) is composed of a control unit and a temperature sensor. The system employs a Wheatstone bridge, a transistorized gating circuit, and an output relay. One leg of the Wheatstone bridge is the temperature sensor. The resistance of this element decreases with increasing temperature. When the average temperature of the sensing element decreases to the value that balances the bridge, the gating circuit operates the output relay. The output relay activates the cockpit annunciator panel structural overheat and master caution warning lights.

The control unit, which is located in the electronic compartment, contains the fixed bridge elements, the gating circuit, and the output relay. The sensing element is a coaxial cable with a stainless steel center conductor and outer conductor; it is filled with a highly compacted dielectric material that has a negative temperature coefficient of resistance. The cable is 103 feet long and follows the fuselage structure adjacent to the nose-fan scrolls, both nose-fan bleed ducts, both wing-fan scrolls, both tail-pipe ducts, and the forward and aft main spars in the vicinity of the wing-fan scrolls. Adjustment of the alarm point is accomplished by varying the value of resistance in another leg of the control unit bridge. Electrically, the sensing element is a single closed loop. Conventional wiring is used to connect the sensing element segments where temperature sensitivity is not desired.

MAINTAINABILITY HISTORY AND RECOMMENDATIONS

Overall Performance

The basic adjustment of the warning system was established primarily from data derived from tests in the Ames Research Center 40- by 80-foot wind tunnel. The system is set to activate the warning light when structural temperatures approach the high-temperature/low-stress limit (Table III). Some modifications of the alarm point and detail routing were made during flight test operations, but no systematic flight testing of system operation and correlation with ground checks of alarm point were performed.

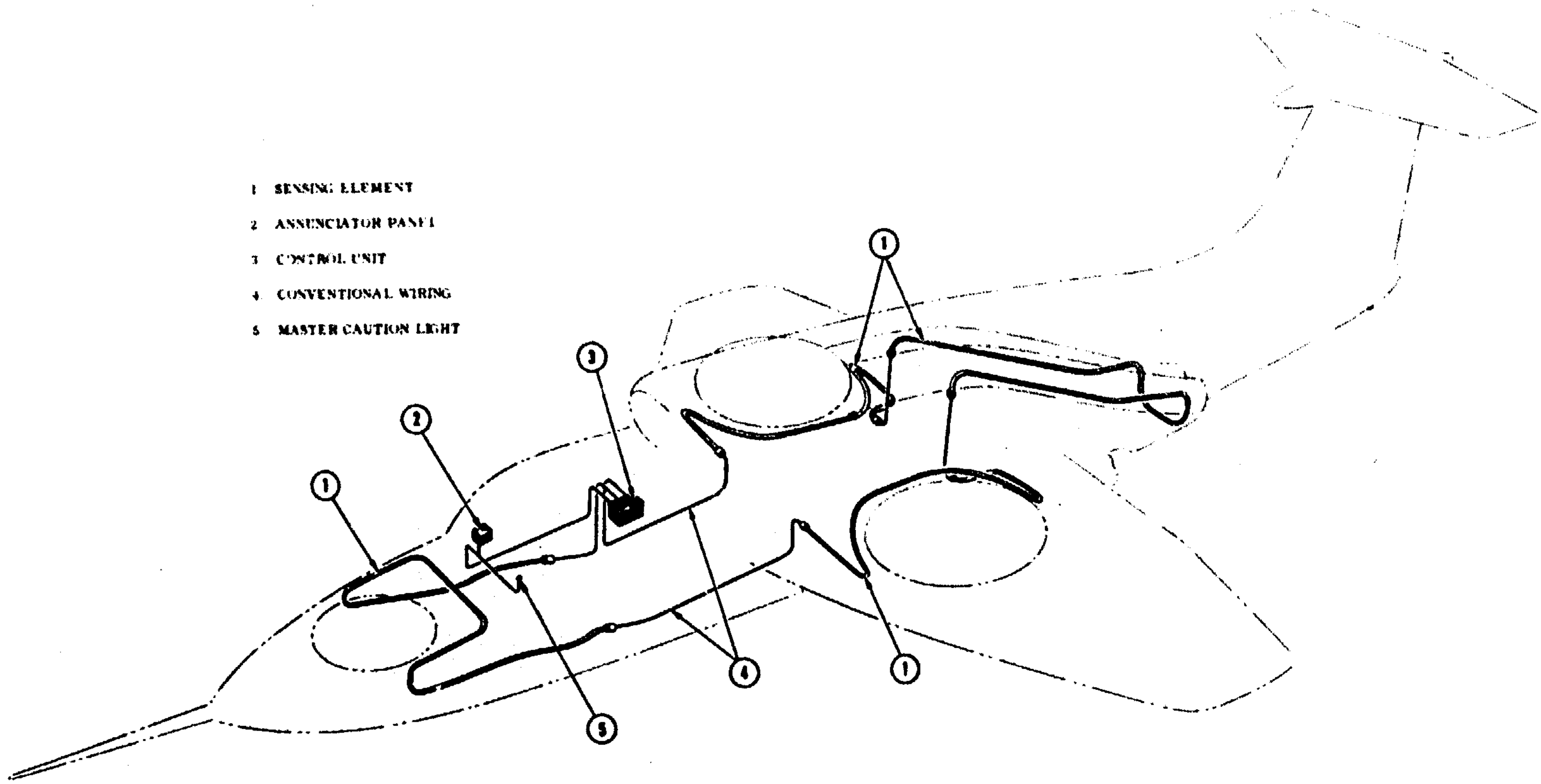


Figure 26. Structural Overheat Warning System Arrangement.

In terms of overall aircraft flight safety, the system has been satisfactory. However, occasional warning activations have occurred. Some of these false warnings were caused by the sensor cable's coming into physical contact with hot gas components. Improved maintenance practices appear to have reduced this problem. Conditions indicating local heating with no overheat warning have also occurred. No major structural damage was incurred. However, for an operational aircraft, where long structural life and minimum maintenance are required, the accumulative effects of undetected low-level structural degradation due to heating must be prevented. A thorough, time-consuming inspection is required in all areas of the aircraft; the system must be monitored to identify and validate the indicated trouble because the sensing element is a single loop.

Sensitivity

The sensing element responds to general heating along its length and to local heating. Both characteristics are desirable for this type of system. However, when these characteristics are coupled with the relatively high allowable operating temperatures, with the rather long length of the sensor required in this system, and with the close proximity of the sensor element and the hot gas components in some areas (due to space limitations), setting the alarm tends to become a sensitive adjustment.

A practical method of reducing the sensitivity would be to divide the system into a set of localized loops (that is, into more discrete segments), each with its own control unit and alarm trip point, as required. All control units would be capable of individually or collectively activating the cockpit annunciator system. To reduce the maintenance downtime for isolating the location of indicated overheating conditions, a simple, manual resetting indicator could be installed for each loop that induces the overheat warning. These indicators could be remotely located in any area convenient for inspection purposes rather than in the cockpit, where space is at a premium. In prototype aircraft, they would be most useful as flight test instrumentation. A well matured aircraft configuration should have little need for such information.

DESIRABLE FEATURES

1. The system provides sensitivity to both localized and general heating conditions in areas monitored.
2. The system is self-restoring after thermal actuation.
3. The system is simple.

UNDESIRABLE FEATURES

1. There is a single sensing loop for all monitored areas.

2. Overheat warnings are sometimes false.

3. Structural overheating conditions are not identified by the warning system.

4. Adjustment of the alarm point is sensitive.

PROPULSION LIFT SYSTEM

SYSTEM CONFIGURATION AND OPERATION

General

The propulsion system consists of two J-85 turbojet engines (less afterburners) used as gas generators, two X353-5B lift fans, one X376 pitch fan, and associated diverter valves. It is a convertible system which augments the thrust of the two turbojet engines for vertical takeoff and landing.

Each of the two J-85 turbojet engines and the lift-fan ducting are pneumatically coupled by a diverter valve. For vertical flight, the diverter valve directs exhaust gases from the turbojets through ducts into the tip turbine scrolls of the two lift fans and the pitch fan. During transition from hover to horizontal flight, louvers located on the lower surface of the fan vector the fan exhaust rearward to provide horizontal thrust for forward acceleration. Once the aircraft has reached a speed sufficient for wing-supported flight, the diverter valve is repositioned to the straight-through position, thereby diverting the hot gases from the wing- and pitch-fan turbines to and through the tail pipes. The wing-fan inlet doors and exit louvers and the pitch-fan inlet louvers and thrust modulator doors are closed, and the J-85 turbojets operate in a conventional manner. The major components are described in the paragraphs that follow.

Diverter Valve Body

With the exception of the actuation system, the diverter valve body (see Figure 27) is constructed from AMS 5536 stainless steel (Hastelloy X) and is basically a cylinder and a constant-radius elbow blended together to form a duct with two branches. The 0.032-inch-thick shell of the body is pierced at two points on each side for door bearings. The two bearings for the forward door are set in die-formed bearing pads located approximately along the centerline of the elbow section. The two rear-door bearings are incorporated in the trunnion mount pad. All four bearings consist of hard Stellite No. 6 bushings pressed into Hastelloy X sleeves that have been welded to the valve body shell and the bearing pad. The bushings are flanged so that they can take radial axial loads; thus, the valve doors can be used as "tie rods" to help stabilize the shell against pressure loads. The average clearance in the bearings is 0.020 inch to protect against seizure. The bearings operate without lubrication at a gas temperature

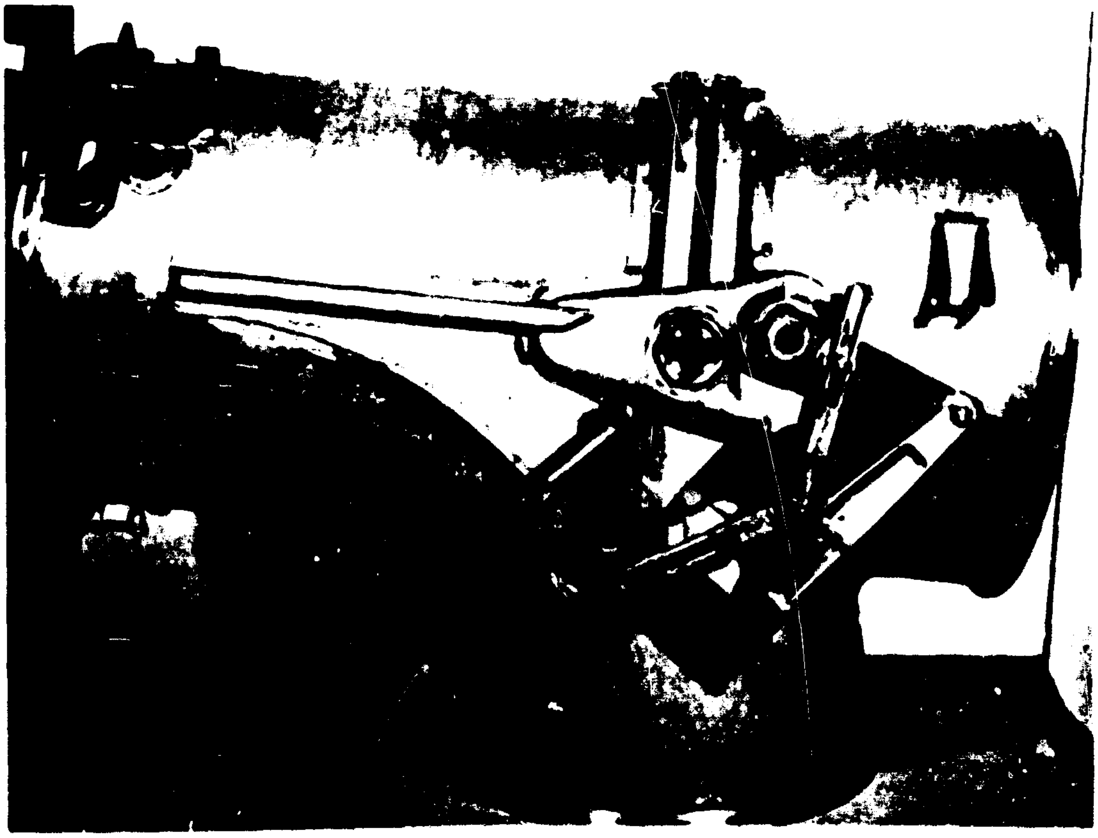


Figure 27. Diverter Valve Assembly.

of 1265° F. A stiffening ring half encircles the body and connects to the two trunnion pads; it has an inverted "W" cross section, and only the outermost ends of the "W" are attached to the valve body to prevent the buildup of thermal stress in the shell.

Diverter Valve Doors

One entire side of each valve door (see Figure 28) is made in a single, continuous, 0.044-inch-thick skin, the central portion of which forms one flange of a hollow-box center beam. Extending fore and aft from the center beam, and welded along the continuous skin, is a series of flanged ribs spaced every 1.5 inches. These are made of 0.030-inch stock in the rear door and of 0.044-inch stock in the forward door. Each end of the center beam encases a stub shaft socket block that is bored to a precise diameter so that the stub shaft that supports the door will fit with a minimum of play. A tapered slot at the full depth of the bore provides the means for transmitting torque into the door from the actuator. At the base of the slot, a self-locking flare nut, permanently installed, is used in fastening the stub shaft in place. The entire rim of the door is formed by a

5/8-inch-diameter, 0.045-inch-wall-thickness seal tube, which provides a housing for door-to-body seal pieces.

During the starting of the engine and until the valve door can achieve an equilibrium temperature, a thermal gradient exists between the surfaces and the center of the door. Without protection, this gradient could be as much as 800° to 900° F. Therefore, the pressure side of each door is protected from radiation or direct hot gas stream impingement by heat shields. The heat shields are separate, unstressed panels that are die-formed in two pieces from 0.015-inch stock, seam-welded together, and held in place on the door by being interlocked with each other and with the flanges of the ribs. Each heat shield is slid into place between two ribs and is secured to the seal tube at one end by tack welds.

The seal is an articulated series of metal pieces retained in the slotted tube at the rim of the doors. The seal pieces are made of die-formed parts brazed together. Each seal piece interlocks with the adjacent one by means of a 1/4-inch-long tailpiece. Sufficient clearance is allowed in the tube to permit the string of seal pieces to curve and twist as required to conform to the sealing contour. Contained within the seal tube and under the seal pieces is a one-piece, canted, helical spring made of 0.020-inch-diameter Inconel X wire. The pressure of the gas holds the seal piece in contact with the valve body. Since the valve doors are non-circular and not perpendicular to the duct centerlines in their closed positions, their edges make varying angles with the duct wall. Therefore, the tube slot is a continuous spiral to make a fixed angle to the wall. The seal tube is interrupted only at the bearing block, where a 1/2-inch-wide gap forms an assembly slot for seal pieces and springs. This 1/2-inch gap is sealed by a three-piece tubular seal, which is held in place by the duct wall when the door is assembled in the diverter valve.

Diverter Valve Actuation System

The diverter valve actuation system (see Figure 28) includes an activating cylinder, connecting linkage for coordinating door motion, stops for limiting door motion, and stub shafts for supporting the doors and transmitting the torque. The actuator is a 3000-psi linear hydraulic component that responds to pilot commands through a pair of aircraft-supplied, four-way, three-position, solenoid hydraulic valves. Reliability requirements necessitate the use of a tandem piston actuator with completely separate and independent hydraulic circuits. Positive actuator cooling is provided by incorporating an 0.015-inch orifice through each piston. Actuators are interchangeable for right- and left-hand diverter valves. They can be operated satisfactorily throughout an ambient temperature range of from 0° to +300° F and throughout a fluid temperature range of from 0° to +300° F; there should be no leakage through static seals. The principal

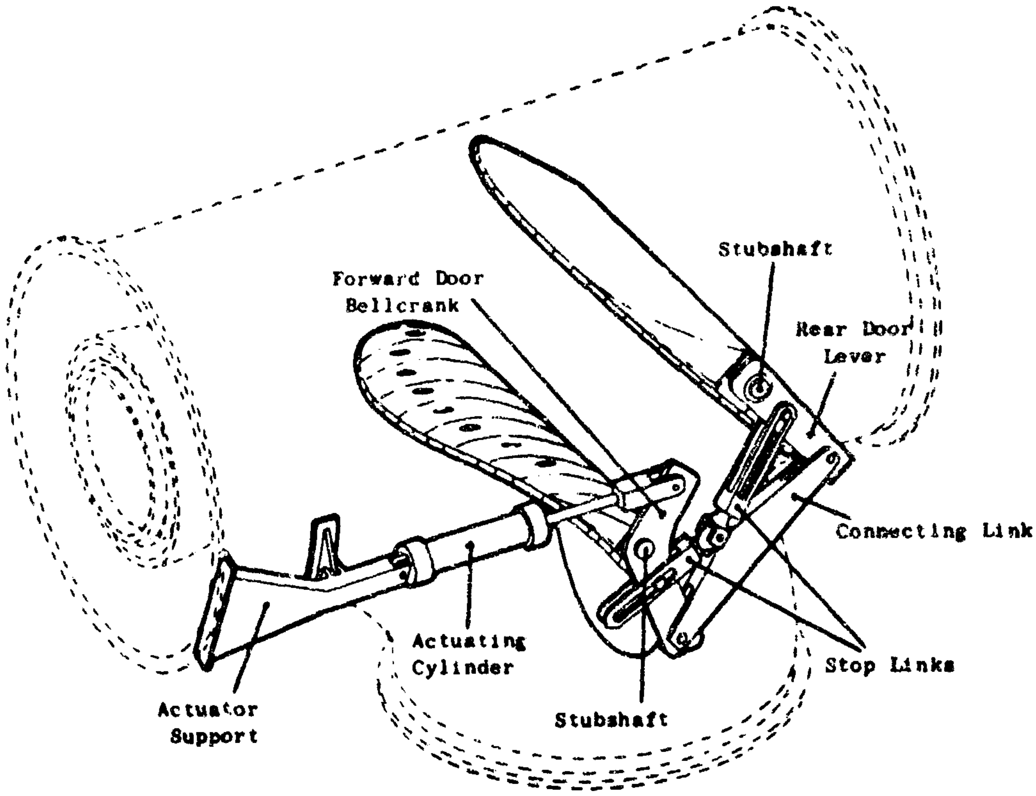


Figure 28. Diverter Valve Actuation System.

parts of the actuation linkage (see Figure 29) are a bell crank mounted on the forward door, a lever mounted on the rear door, and a connecting link between them. About midway along the connecting link is a pin which provides a pivot for two adjustable stop links. The other ends of the stop links are slotted and slide on pins on the bell crank and lever. Thus, the linkage motion is stopped when either stop link reaches the end of its travel on the pin.

Mechanical Coupling Between Diverter Valves

A mechanical coupling connects the two diverter valves in the system to ensure simultaneous operation in the event of a complete actuator failure on one valve. Unless both valves have the same mode setting (straight through or diverted), a large yawing moment will be imposed on the aircraft from the jet thrust deflectors employed in the control system. The coupling is located between the aft doors of each valve; it uses the doors for transmitting the actuation torque. During normal operation, it rotates freely with the doors without transmitting any torque or imposing any stresses in the valves. Free thermal expansion is also permitted, and the telescopic action of the torque shaft allows assembly without affecting

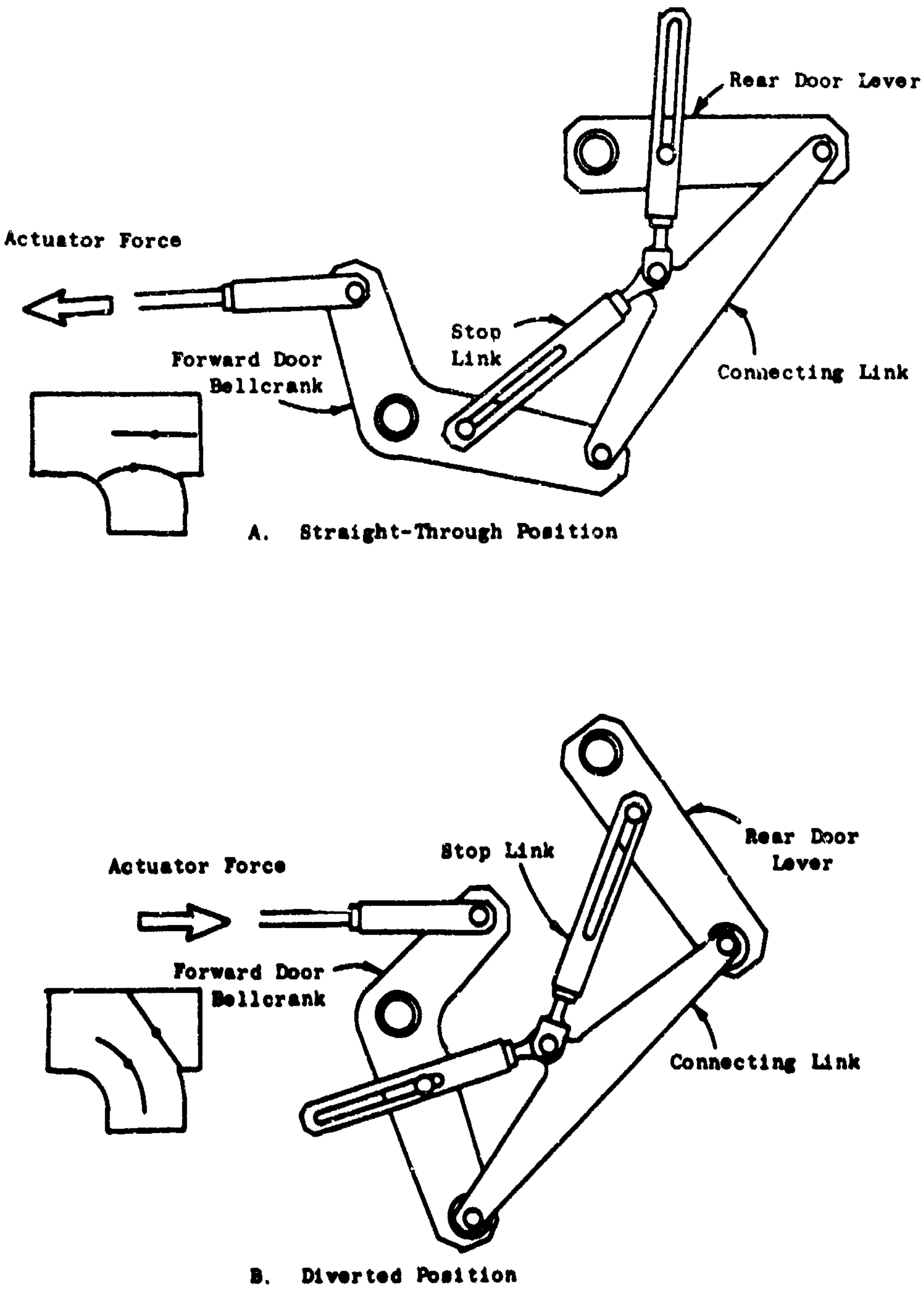


Figure 29. Diverter Valve Actuation Linkage.

the valve installation. The torque shaft is constructed in two pieces and uses eight radial teeth, four equally spaced on each end, for transmitting the torque. These teeth engage in radial slots in the door bushings on each valve. Angular misalignment of the two doors is accommodated by match-drilling the two-piece torque shaft for a radial shear bolt.

Wing Lift Fans

The two wing lift fans are located immediately adjacent to the engine diverter valve coupled by the crossover ducts. The lift-fan assemblies consist of inlet guide vanes, a front frame, a scroll, a rotor rear frame, and variable exit louvers. The two rotors, as mounted in the aircraft, rotate counter to each other to eliminate undesirable gyroscopic coupling. The lift-fan rotor component and the lift-fan front frame are shown in Figures 30 and 31, respectively.

Each rotor has 36 blade platforms, which form the inner aerodynamic flow path through the fan. The platforms and the stiffening braces are formed of 0.028-inch 615 aluminum. The assembly is spot-welded, brazed, and heat-treated. Two bolt-hole grommets and washers are swaged to the platform assembly. The platform assemblies are bolted to the tabs on the retainer rings.

The scroll (see Figures 32 and 33) accepts exhaust gas from two gas generators and ducts the gas through two "arms" to the 167.5^{o} arc of the nozzle diaphragm. The scroll has two inlets; each inlet is connected through cross ducting to both J-85 gas generators. Thus, each scroll inlet is designed to accept 50 percent of the flow of each gas generator. The cross-ducting arrangement separates the flow from each of the engines and permits the fans to continue to operate at part speed in the event that one gas generator shuts down.

The nozzle assembly of each wing fan has a total of 13 adjustable vanes, which are assembled between the partitions near the ends of the scroll arms. These vanes are adjusted to trim the flow of hot gas through the nozzle diaphragm (see Figure 34). Five adjustable vanes are located in one arm near the 12 o'clock position; eight are in the other arm near the 6 o'clock position. An adjusting shaft for each vane extends from the outer side of the scroll and passes through the boss and sleeve to the end of the vane. A slot on the end of the shaft engages the end of the vane to maintain the angular setting. A nut with external threads retains the shaft axially but permits angular rotation. A lever arm attached to the end of the shaft holds the shaft at any one of four positions.

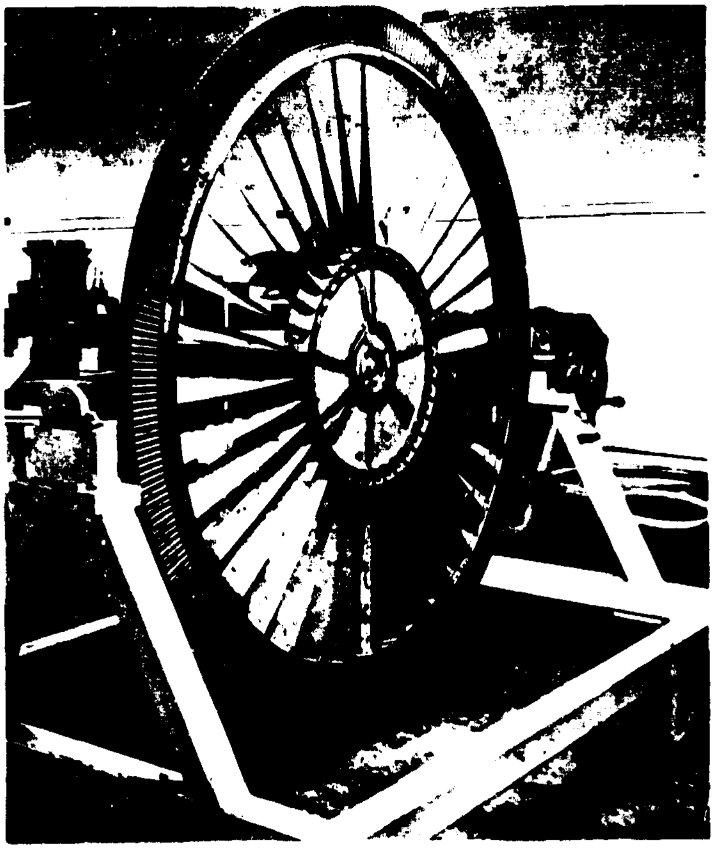


Figure 30. Lift-Fan Rotor Component.

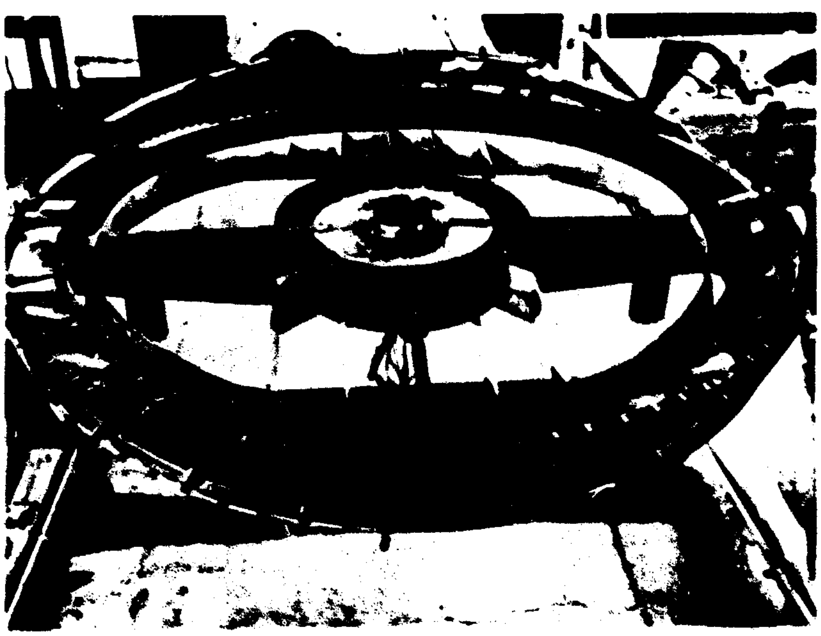


Figure 31. Lift-Fan Front Frame.

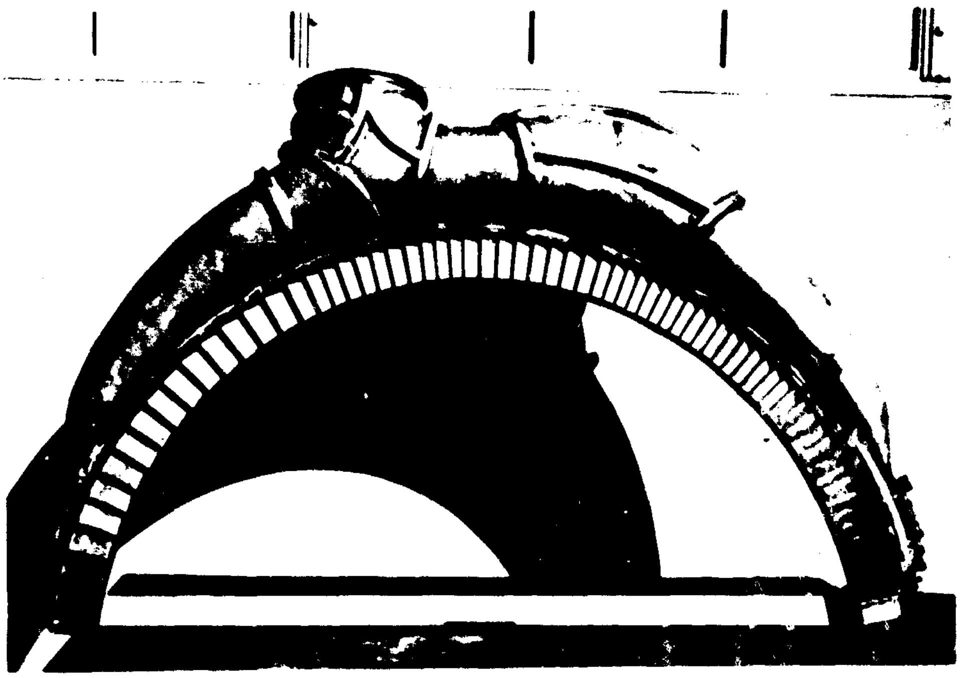


Figure 32. Lift-Fan Scroll, View of Nozzle.

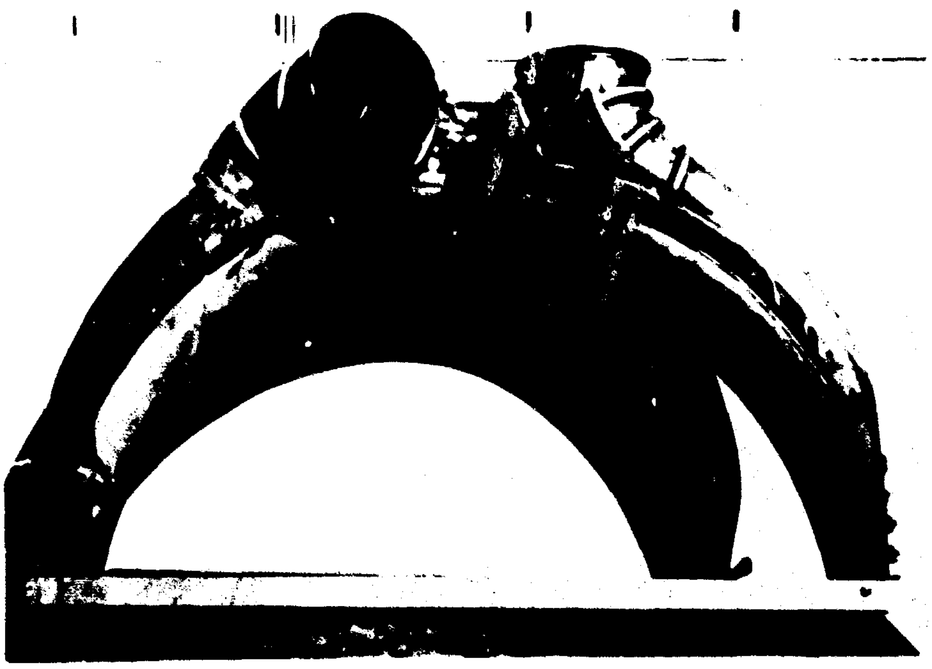


Figure 33. Lift-Fan Scroll, Top View.

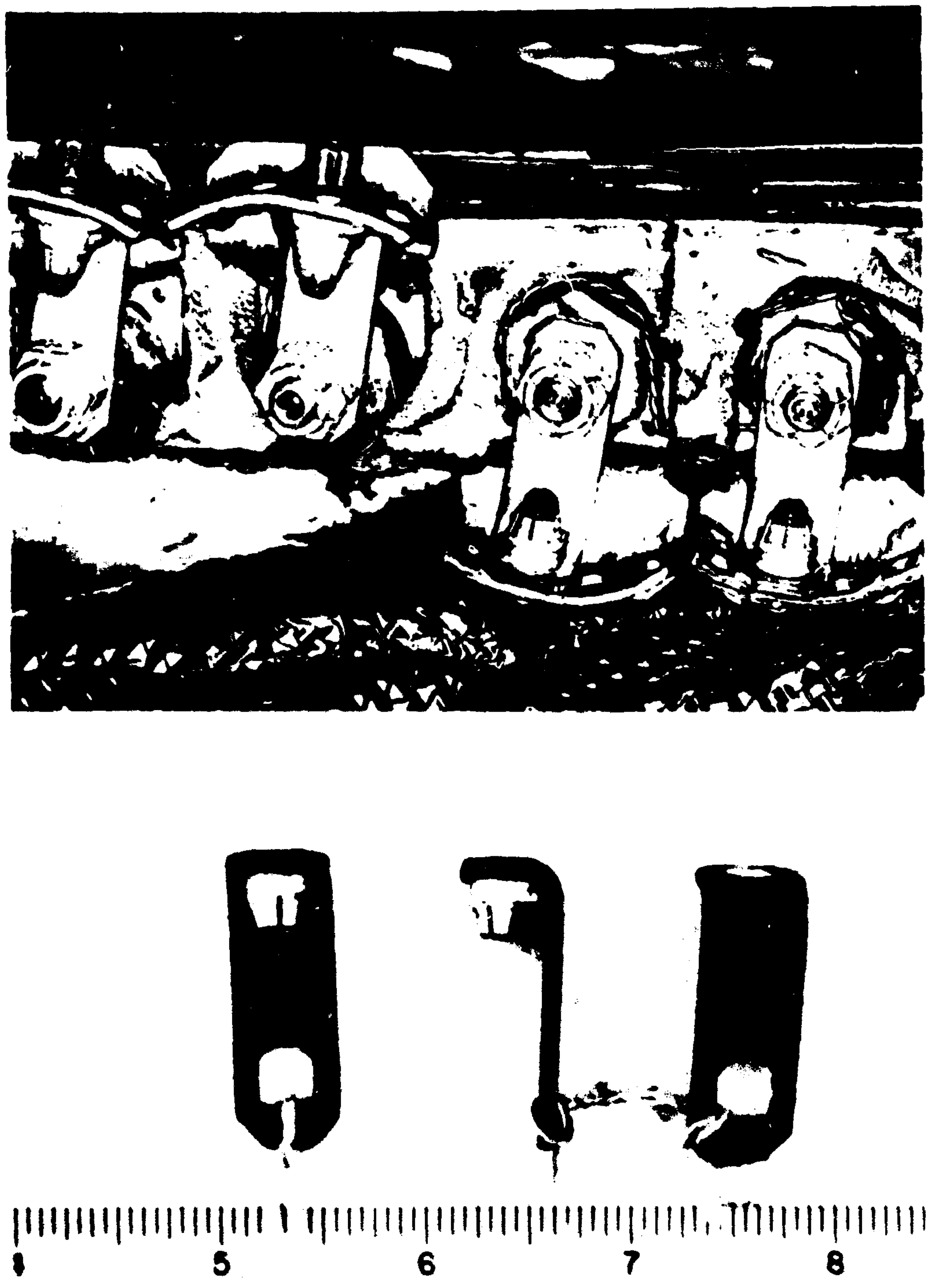


Figure 34. Scroll Area Adjustment Mechanism.

A fuel drain at the lowest point in the scroll provides a means of removing any raw fuel that escapes because of an engine false start, thus precluding the possibility of the fuel's burning in the scroll assembly.

Pitch Fan

The pitch fan is aerodynamically similar to the wing fans, but it incorporates a more advanced lightweight mechanical design. Pitch control in fan-supported flight is effected through modulation of the exhaust flow direction from the pitch fan. In wing-supported flight, the thrust modulator doors and pitch-fan inlet louvers close to form an integral part of the fuselage, thus resulting in an aerodynamically clean airplane for the conventional flight mode. The pitch fan also provides 700 to 800 pounds of vertical thrust for the hover mode.

The input power represents a constant-percent bleed of the gas generator turbine discharge, the level of which is established by adjustment of the nozzle area in the pitch-fan gas inlet scrolls. The ducting that connects the gas generators and the scrolls is provided as part of the airframe; if desired, it can incorporate special bleed valving to shut down the pitch fan during the VTOL flight mode.

Scroll blank-off plates (see Figure 35) are assembled, as required, inside the scroll during initial assembly to reduce the nozzle area and to control the flow of hot gas to a desirable level. The blank-off plates are cup shaped, with the outer contour conforming to the shape of the scroll. Three lugs welded inside the cup are match-drilled to the bosses in the outer skin of the scroll. A fourth lug welded in the center of the cup is used for assembly purposes (see Figure 36).

Yaw, Roll, and Attitude

Control of yaw, roll, and attitude in fan-supported flight below control surface aerodynamically effective speeds is achieved through a system of differentially variable exit louvers on the bottom of the wing fans. The controls from the cockpit to the louvers are coupled through a mechanical mixer; this arrangement permits the use of conventional stick and rudder controls for both fan-supported and wing-supported flight. The "butterfly" doors close to form the top surface of the wing, and the exit louvers close to form the bottom surface of the wing in wing-supported flight.

Insulation Blanket

The nose-pitch and wing-fan scrolls and the hot gas ducts are covered with an insulation blanket, which consists of a layer of thermal conductance filler material between inner and outer insulated sheet-type layers. The

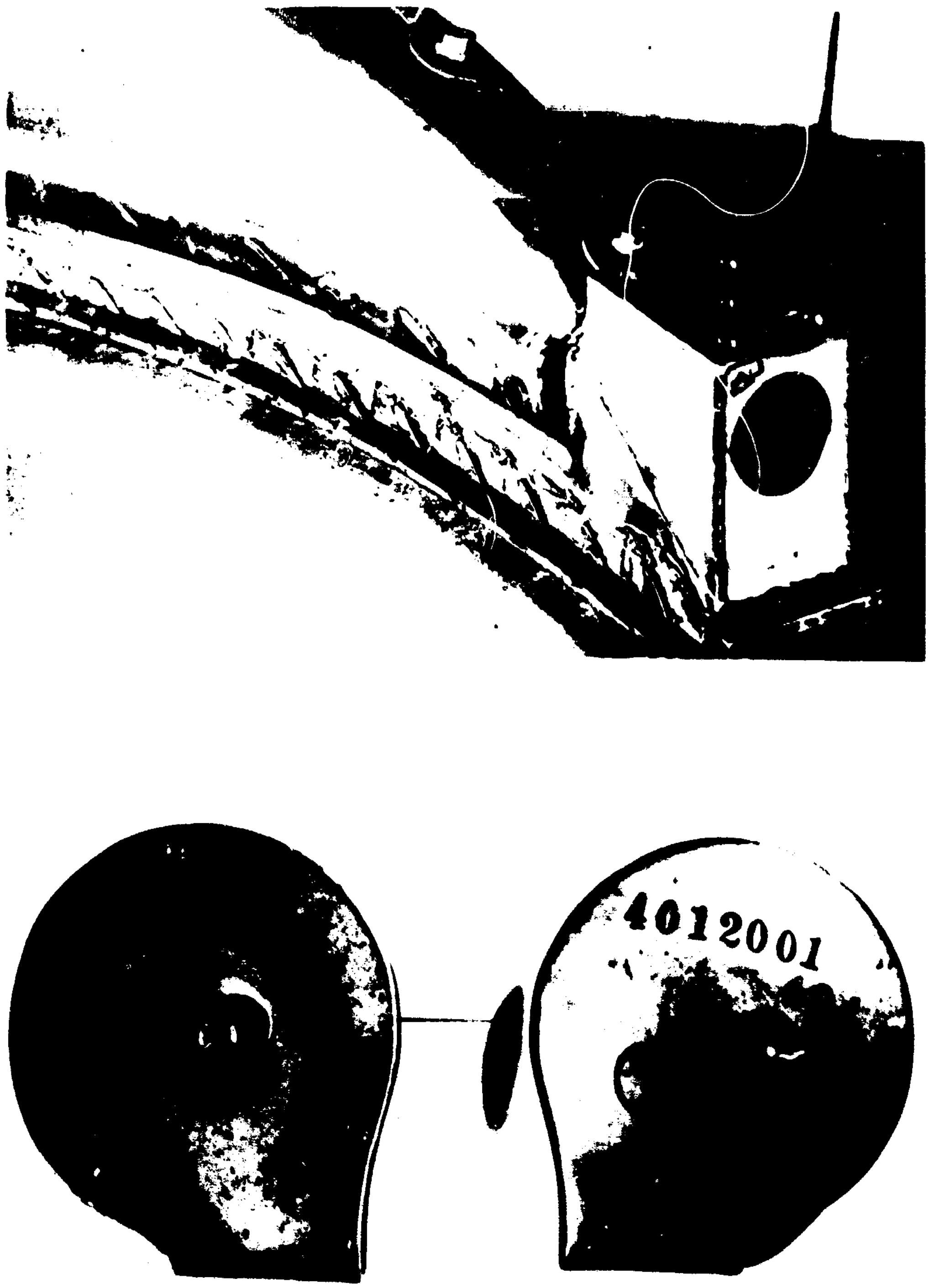


Figure 35. Pitch-Fan Turbine Inlet Scroll.

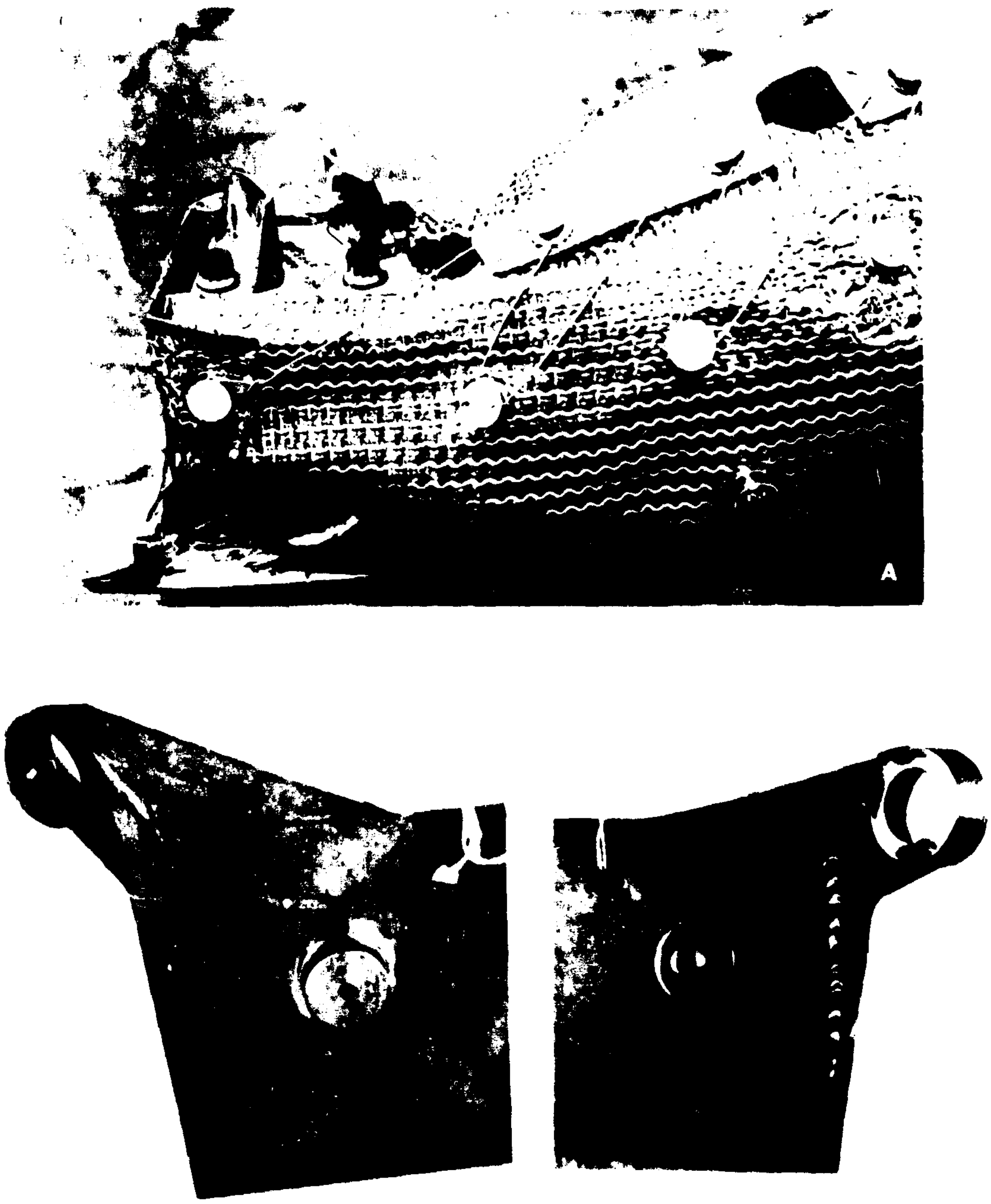


Figure 36. Pitch-Fan Turbine Inlet Scroll, End Mount Clevis, and Support Arm.

blankets are made in sections which contain buttons swaged to the outside skin for lockwire attachment.

MAINTAINABILITY HISTORY AND RECOMMENDATIONS

Failure Rate

Sixty-one propulsion lift-system discrepancies were recorded. The man-hours that were expended in correcting the discrepancies are shown in Table VII. Four of the discrepancies were analyzed and were considered to require further study. They are listed below in order of being the most troublesome to maintain.

TABLE VII. MAN-HOURS EXPENDED IN CORRECTING DISCREPANCIES IN PROPULSION LIFT SYSTEM

Part Name	Preparation	Diagnosis	Accessibility	Corrective Action	Reassembly	Check-out	Total Man-Hours	Remarks
Fan Modification Skin	0.1	0.1	0.1	0.5	0.1	0	0.9	Patched and stop-drilled
L. H. Diverter Valve Door	0.2	0.1	2.0	4.0	2.0	0.2	8.5	Cracks repaired
R. H. Diverter Valve Door	0.2	0.1	2.0	4.0	2.0	0.2	8.5	Cracks repaired
Mixed Valve, J-85	0.3	1.0	0.5	33.0	0.5	0.7	36.0	Assembly replaced
R. H. Diverter Valve Curved Door	0.1	4.0	24.0	11.0	8.0	10.0	57.1	New torque plug installed
L. H. Wing-Fan Pressure Plate	0.1	0.1	0.1	0.8	0.1	0	1.2	Crack repaired
J-85 Engine							32/8*	Removed and re-installed engine
Diverter Valve Doors (4)							16/4	Doors removed and reinstalled after close inspection, with engines already removed

*When 2 numbers are divided by a slash, the first number represents the man-hours; the second, the clock hours.

Diverter Valves

Cracks occurred frequently in the trunnion area, seal tube, and heat shields. If these cracks were located in the forward door and repairs could not be made without removing the door, the engine had to be removed. There was always some leakage around the seals, which caused the temperatures in the wing- and pitch-fan cavities to be near or above limits when the aircraft was flying in the conventional (jet) mode; this was

a contributing degradation factor on certain components. The problems mentioned necessitated an inspection every 5 hours, at which time welding, seal segments rework, or adjustment was required.

Engine Installation

Since the accessory gearbox, which drives the cooling fans and generator, is mounted in a fixed position, it was very difficult to align the drive shaft with the engine without a resulting contact between the engine and structures, which, in turn, caused high engine vibrations. The engines should be rigged in position; then the gearbox should have the flexibility to align with the engine, not the engine with the gearbox.

Wing-Fan Rotor Blade Platforms

The blade platform tabs constantly cracked and broke off.

Wing-Fan Scroll Area Adjustment

This adjustment was required to obtain the proper engine exhaust gas temperature and wing-fan rpm during operations in the fan mode. Accessibility was very poor and adjustment was very time-consuming. An area on the bottom of the wing, the "waffle plate", had to be removed; when it was reinstalled, the sealant that was used (a vulcanized fiber) required 10 hours of curing time before flying. On several occasions, adjustments were made incorrectly because the adjustment mechanism was moved in the wrong direction (see Figure 34).

DESIRABLE FEATURES

1. The actuator on each diverter valve is capable of operating both valves simultaneously through a mechanical link.

2. The two wing-fan rotors rotate counter to each other, to eliminate undesirable gyroscopic coupling.

3. The wing-fan inlet doors and exit louvers and the pitch-fan inlet louvers and thrust modulation doors form an integral part of the fuselage; this results in an aerodynamically clean airplane for the conventional flight mode.

4. The same engines provide the energy required for both fan-mode and conventional-mode flight.

UNDESIRABLE FEATURES

1. Accessibility to engine accessories and associated shafting is limited, and the accessories and shafting are difficult to adjust, install, or inspect.

2. Engines have to be removed every 30 flight hours to drain the engine oil and to clean the oil and fuel filters.

3. Engines have to be removed in order to remove the forward door of the diverter valve assembly.

4. The heat radiated from the hot gas ducts during fan-mode operations exceeds the capability of the cooling system. Thus, desirable internal fuselage temperatures cannot be maintained.

5. The wing-fan doors have no cockpit indicator, nor is there any means of inspecting the electrical actuator locking device to assure that the doors are locked when in the closed position.

PROPULSION SUBSYSTEMS

SYSTEMS CONFIGURATION AND OPERATION

Engine Mounting

The two J-85 turbojet engines, with diverter valves, are located above the wing and aft of the crew station. They are mounted side-by-side in a common nacelle. The engine mounting system consists of master mounts at the diverter valves and vertical mounts at the forward ends of the engines. Side mounts are provided at the lower, forward sections of the diverter valves (see Figure 37).

Common Engine Air Inlet

The engines have a common induction inlet with an internal flow splitter. The inlet is located above and aft of the cockpit canopy. It is provided with a boundary layer bleed duct, which is made of reinforced fiber glass and is removable for servicing engine controls.

Engine Bay

The engine bay is sealed at the aft end by a vertical aluminum fire wall and at the bottom by a horizontal titanium fire wall. The lower fire wall contains holes to accommodate the diverter valves and engine starter lines. Finger seals are used to seal holes around the diverter valves and the engine starter lines. Within the common nacelle, the engines are isolated from each other by a vertical titanium fire wall that runs the length of the engine bay. The top of the vertical fire wall is sealed to the engine compartment top panel with a fire-resistant seal. A vertical fire wall is also included to isolate each engine burner section from its compressor section. A drain system carries away combustible fluids from the engines and ducting and disposes of them below the aircraft.

The engine bay canopy and side access doors are removable so that the engines can be serviced simultaneously. The canopy and doors are an integral part of the fuselage structure, but they do not contain any primary load-carrying members. The panels are constructed of honeycomb core and aluminum skins.

Wing-Fan Mounting

The wing-fan mounting system consists of three mounts for each fan: a forward master mount, an inboard side mount, and an aft fan mount. The forward master mount, which is attached to the forward wing spar, is a ball-and-socket type, capable of accepting loads in all directions. The inboard side mount, which is located at the spanwise centerline of the fuselage, is capable of accepting loads in the vertical, fore, and aft planes. The aft fan mount, which is attached to the aft wing spar, is capable of accepting loads in the vertical and lateral planes (see Figure 37). Flexible seals in the fan-wing surface joints permit relative motion between the fan and wings.

Nose-Fan (Pitch-Fan) Mounting

The nose-fan mounting system consists of a master mount and two side mounts. The master mount is located at the aft section of the fan and is attached to a cantilever trussed structure. The two side mounts are attached to the fuselage longerons (see Figure 37).

Exhaust Ducts

The two tail pipes run diagonally through the aft fuselage (see Figure 38). They are connected to the diverter valves by the tail-pipe flex section. The flex section uses double bellows and provides a semiuniversal joint to accommodate tail-pipe motion due to thermal expansion. Each tail pipe is shrouded for its full length by a titanium tube, which forms an annular cooling airflow passage. The inside of each shroud is gold plated to reduce the transfer of radiant heat to the shroud. The tail-pipe ejectors, one located at the aft end of each tail pipe, consist of conical extensions of the shrouds past the tail-pipe nozzles. The ejectors are effective only during jet-mode operation, when they serve to augment cooling airflow through the engine bay and tail-pipe annuli.

Wing- and Nose-Fan Divider Ducts

The gas power distribution system is designed to provide balanced and proportioned power to the wing and nose fans from each gas generator. In fan-mode operation, the gas is distributed through ducts constructed of stainless steel. The left divider duct (looking forward) supplies the left scroll quadrant of the nose fan and the forward inboard scroll quadrants of the two wing fans. The right divider duct supplies the right scroll quadrant of the nose fan and the aft inboard scroll quadrants of the wing fans. The divider ducts are covered with foil-clad insulation blankets to reduce both hot gas thermal losses and area cooling requirements.

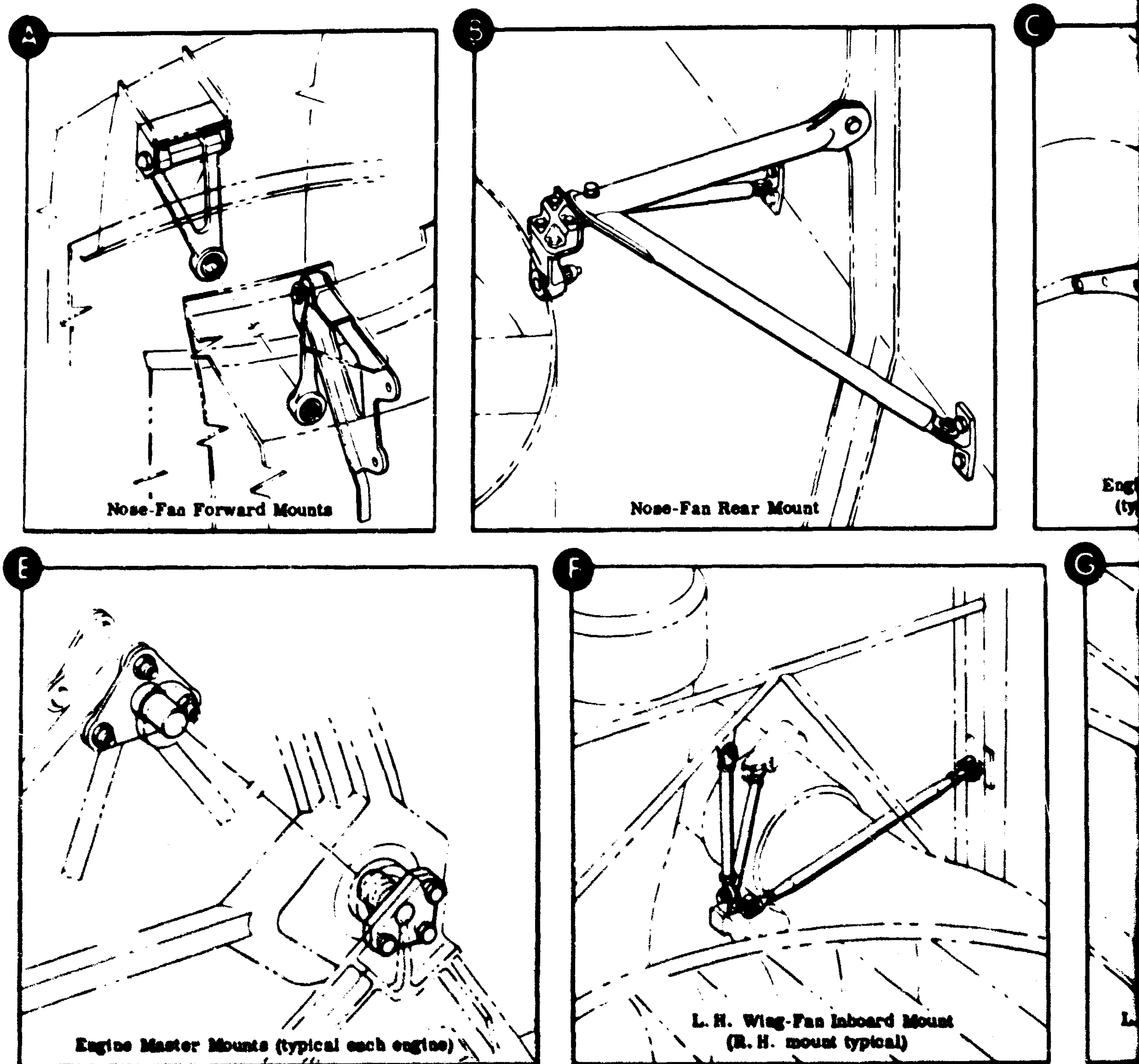
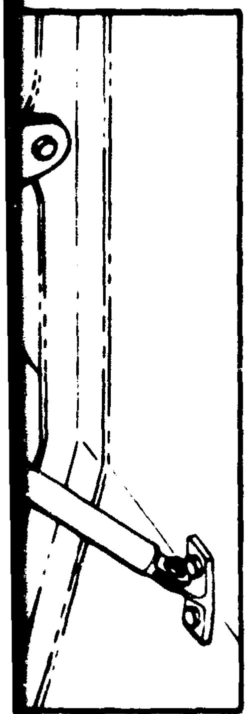
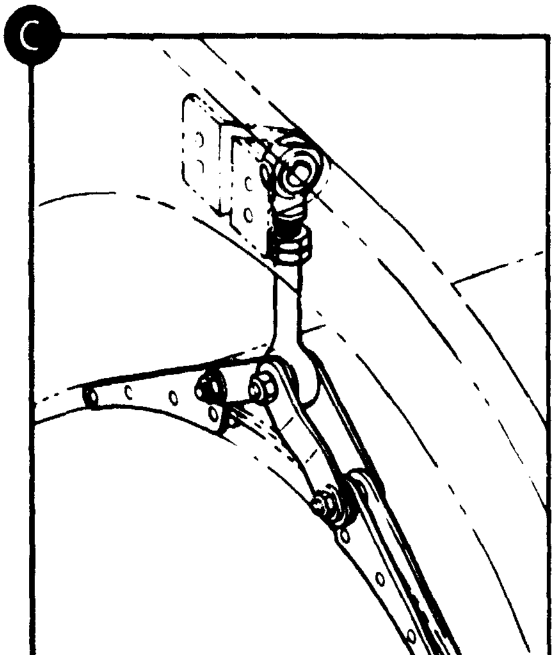
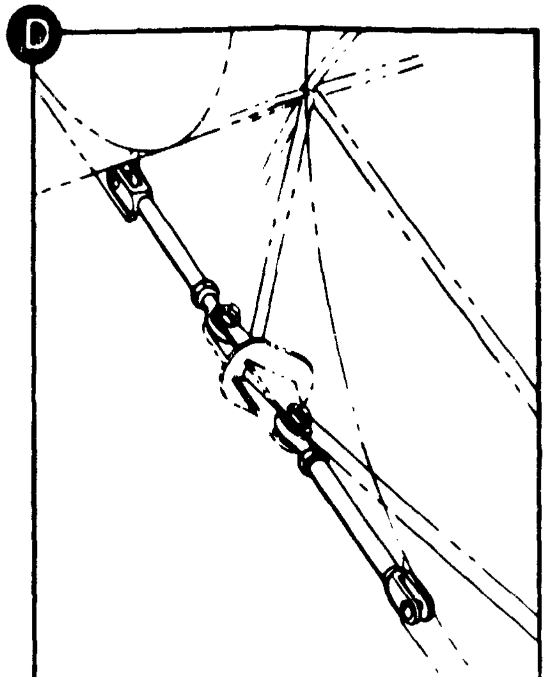


Figure 37. Propulsion System Mounting.

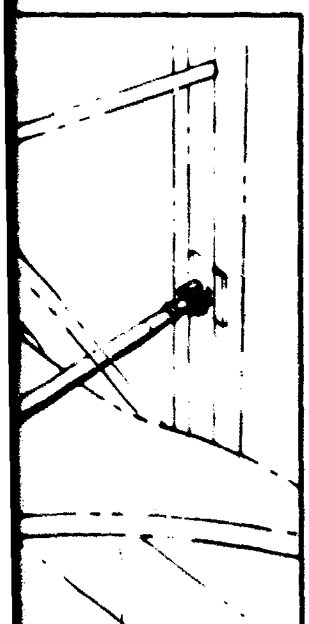




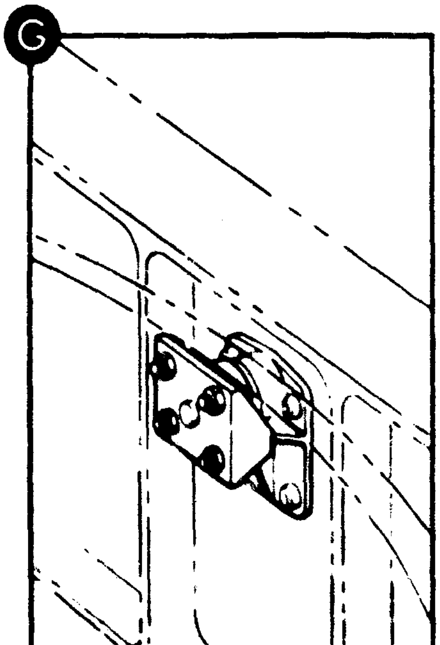
Engine Forward Mount
(typical each engine)



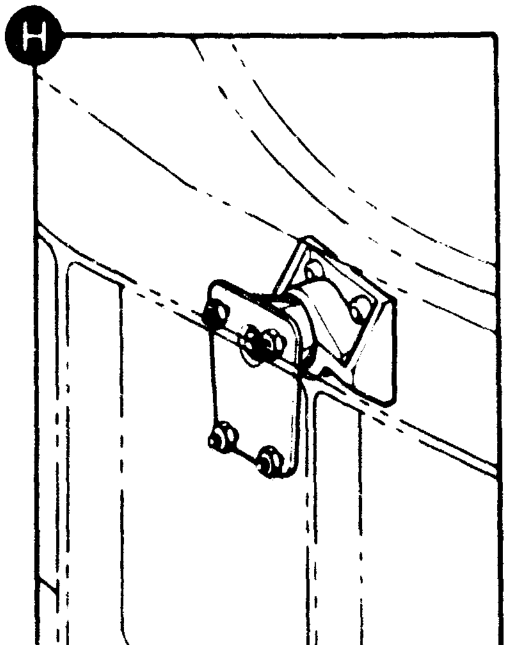
Engine Lateral Mounts



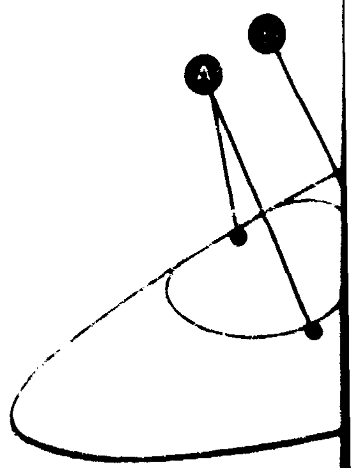
board Mount
ypical)

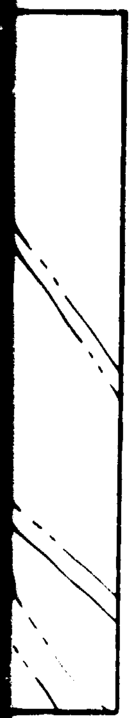


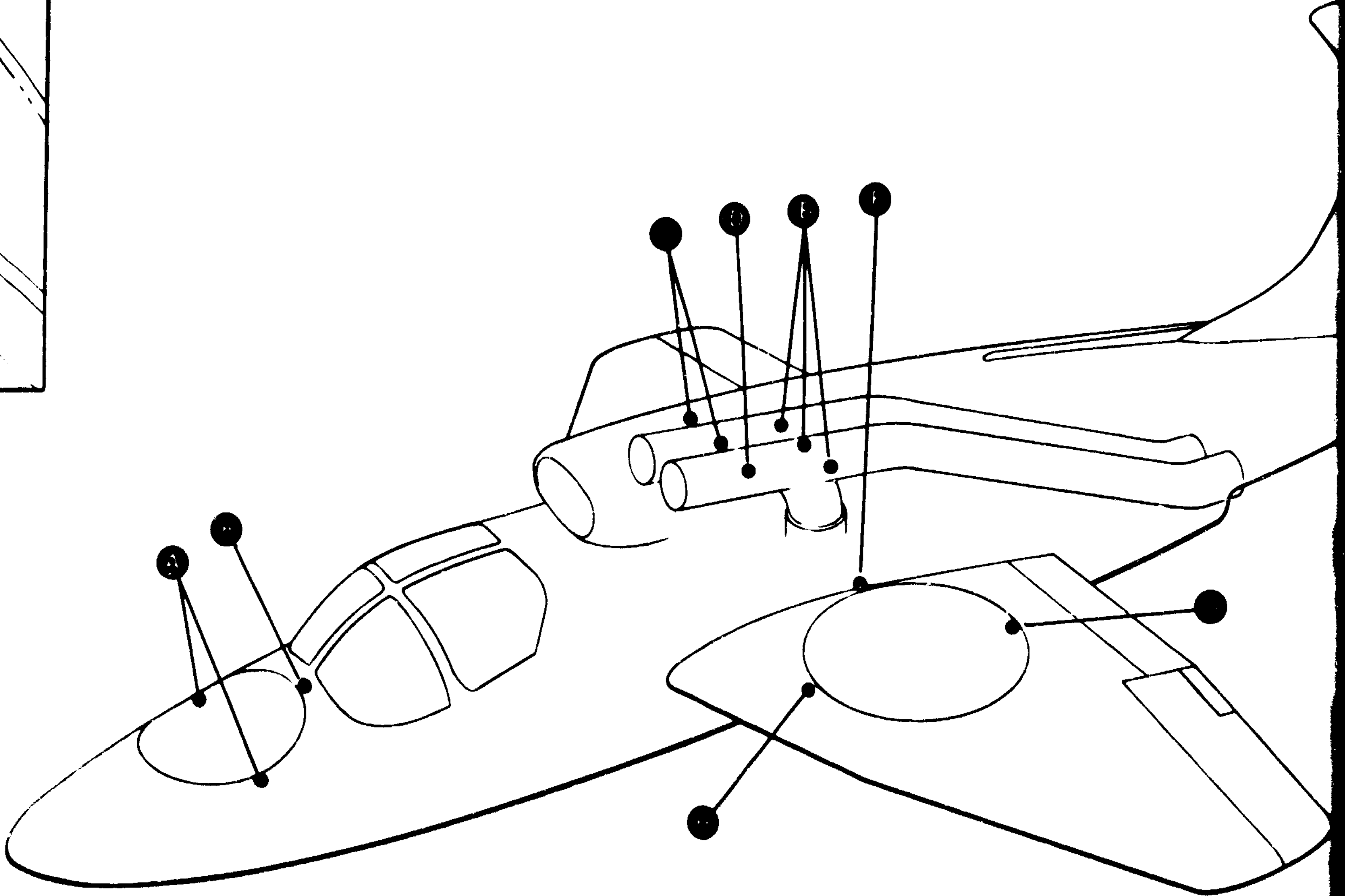
L. H. Wing-Fan Aft Mount
(R. H. mount typical)



L. H. Wing-Fan Forward Mount
(R. H. mount typical)







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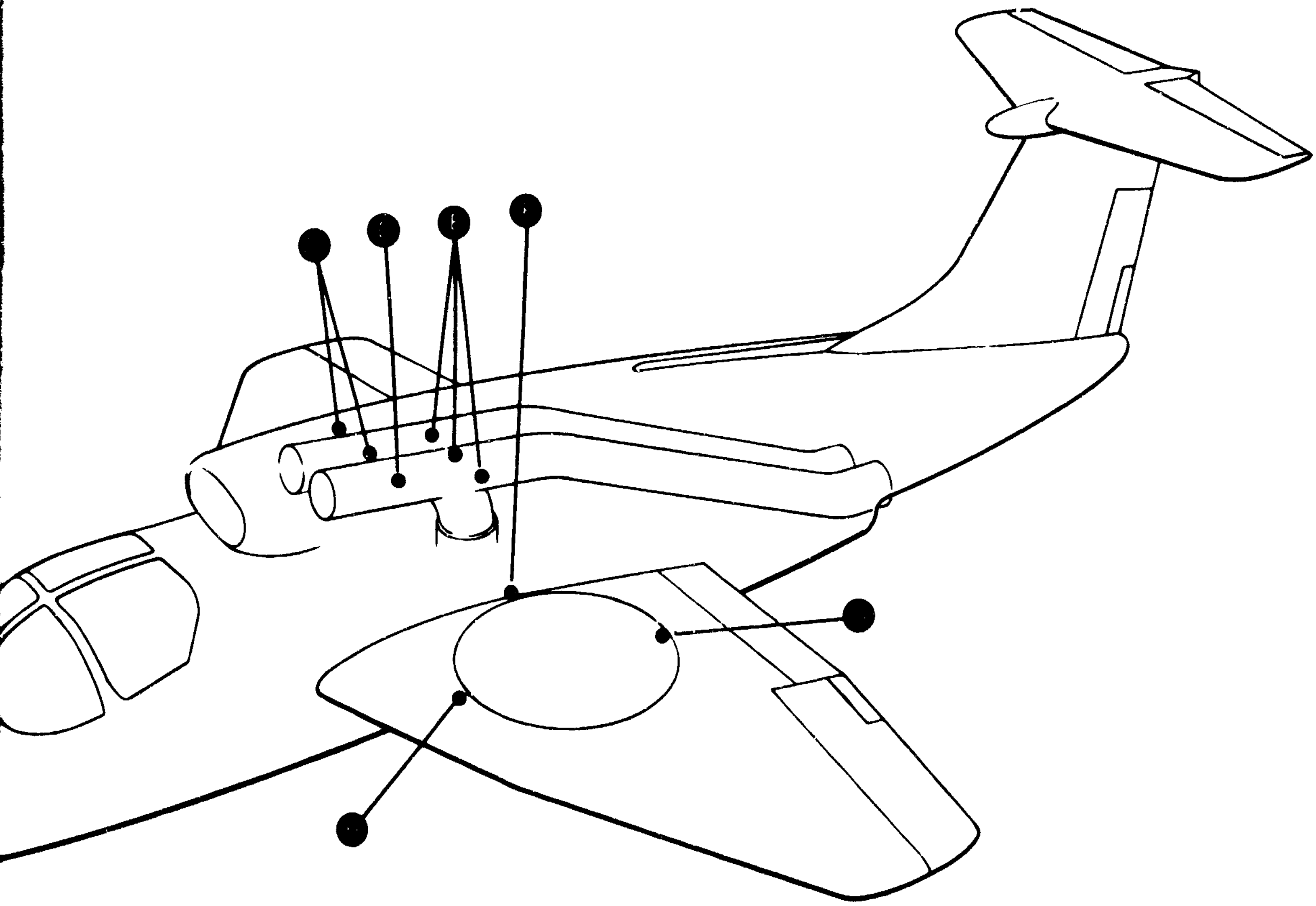

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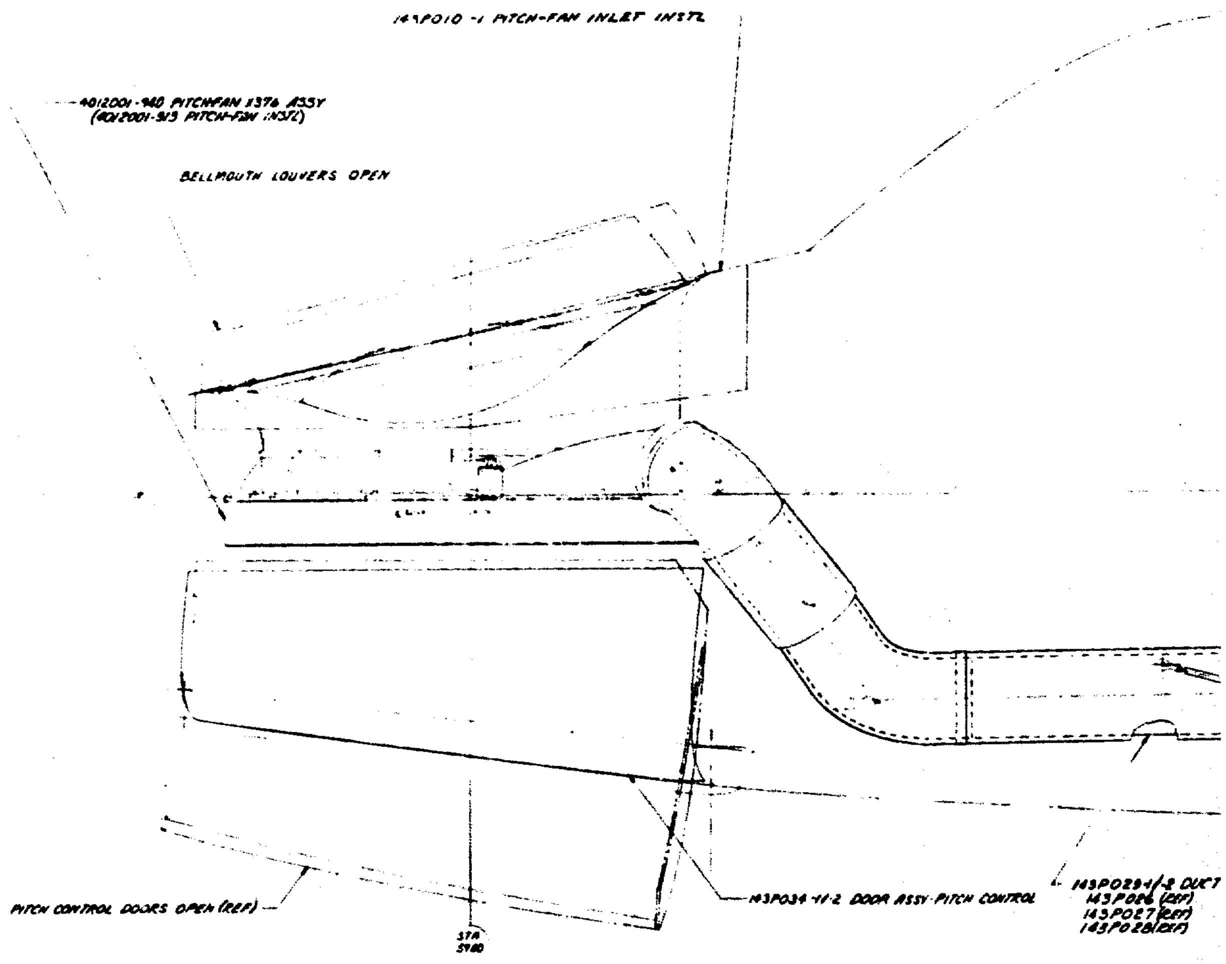

Figure 38. Propulsion System Installation.

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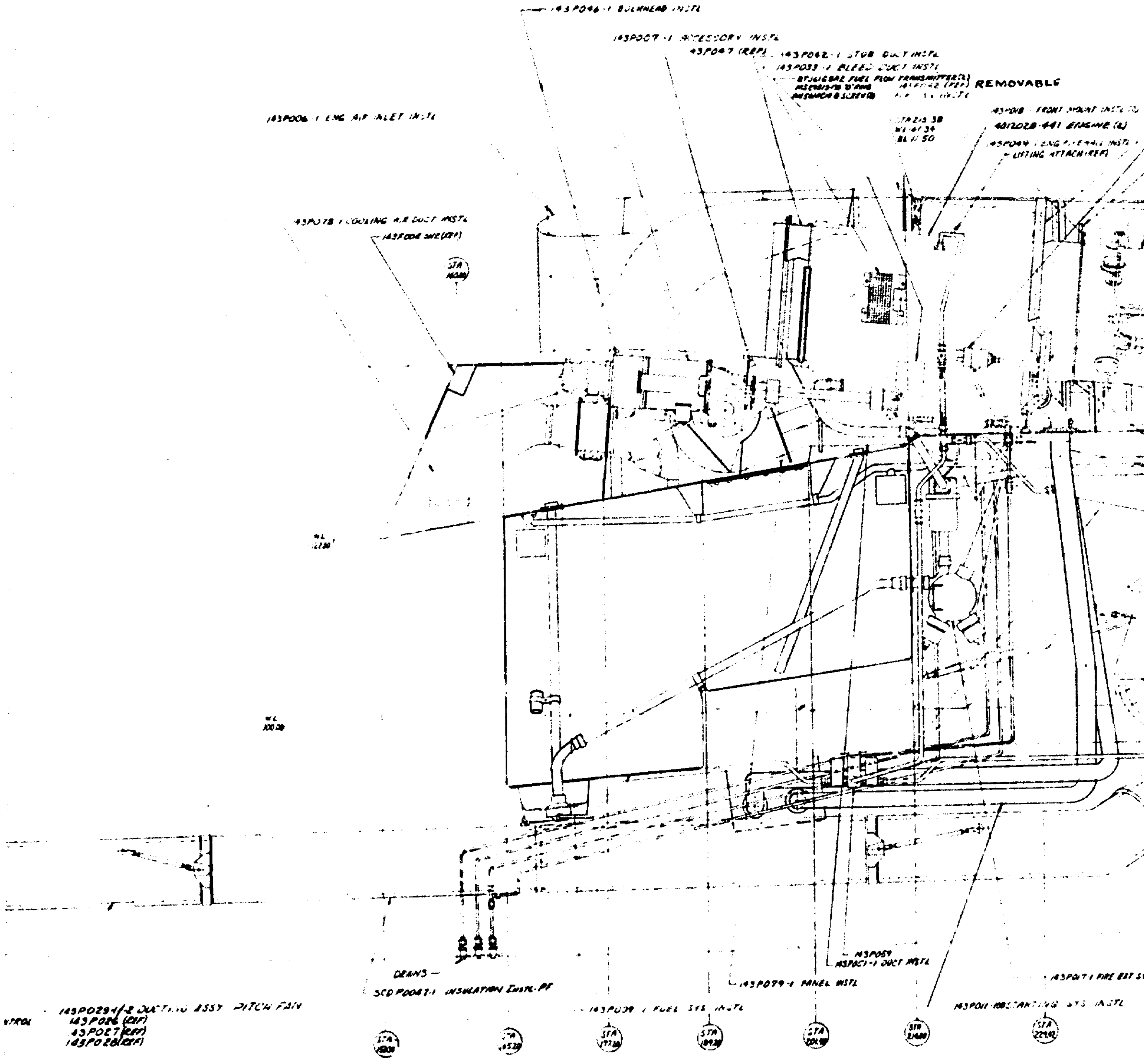
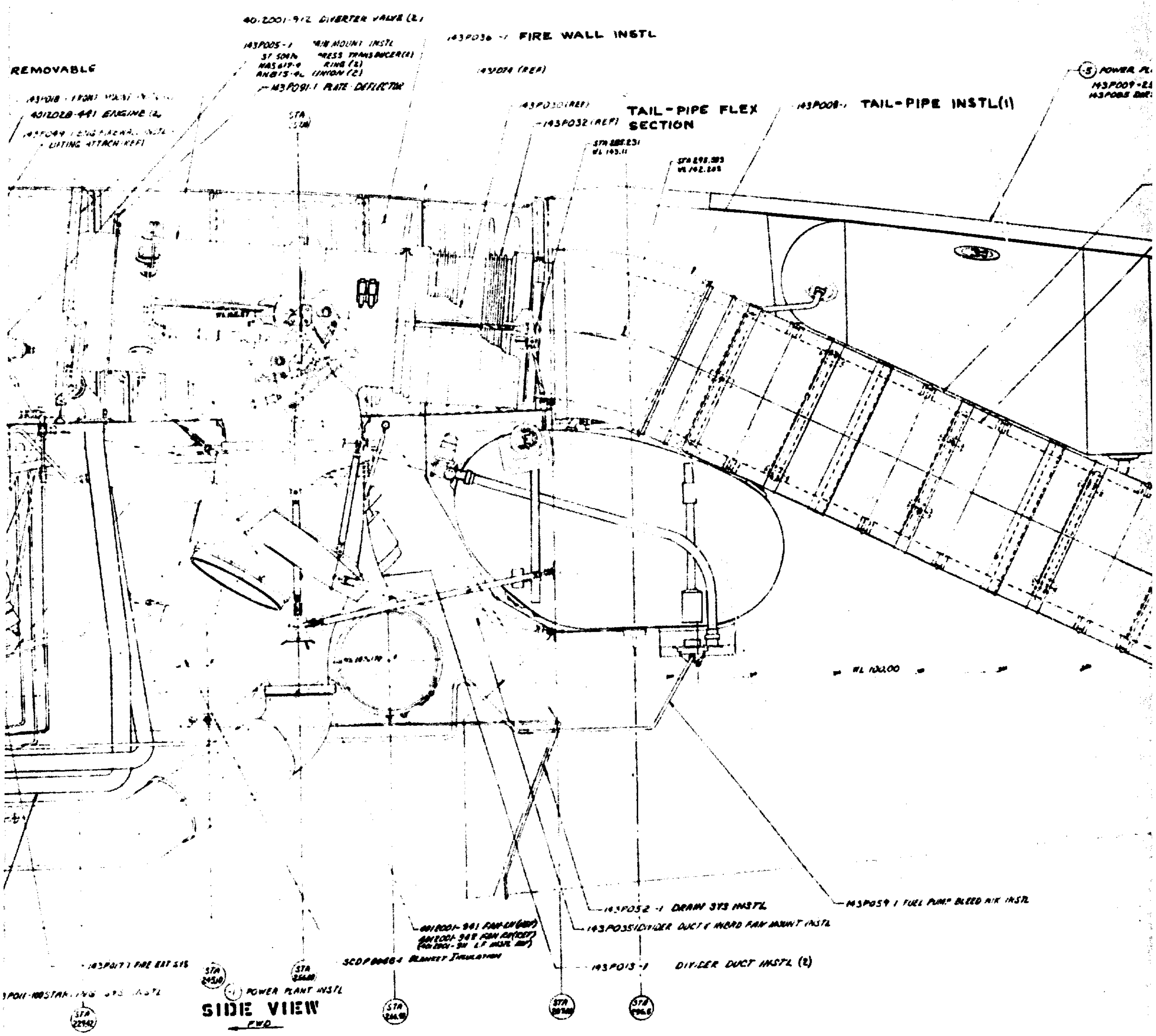
A



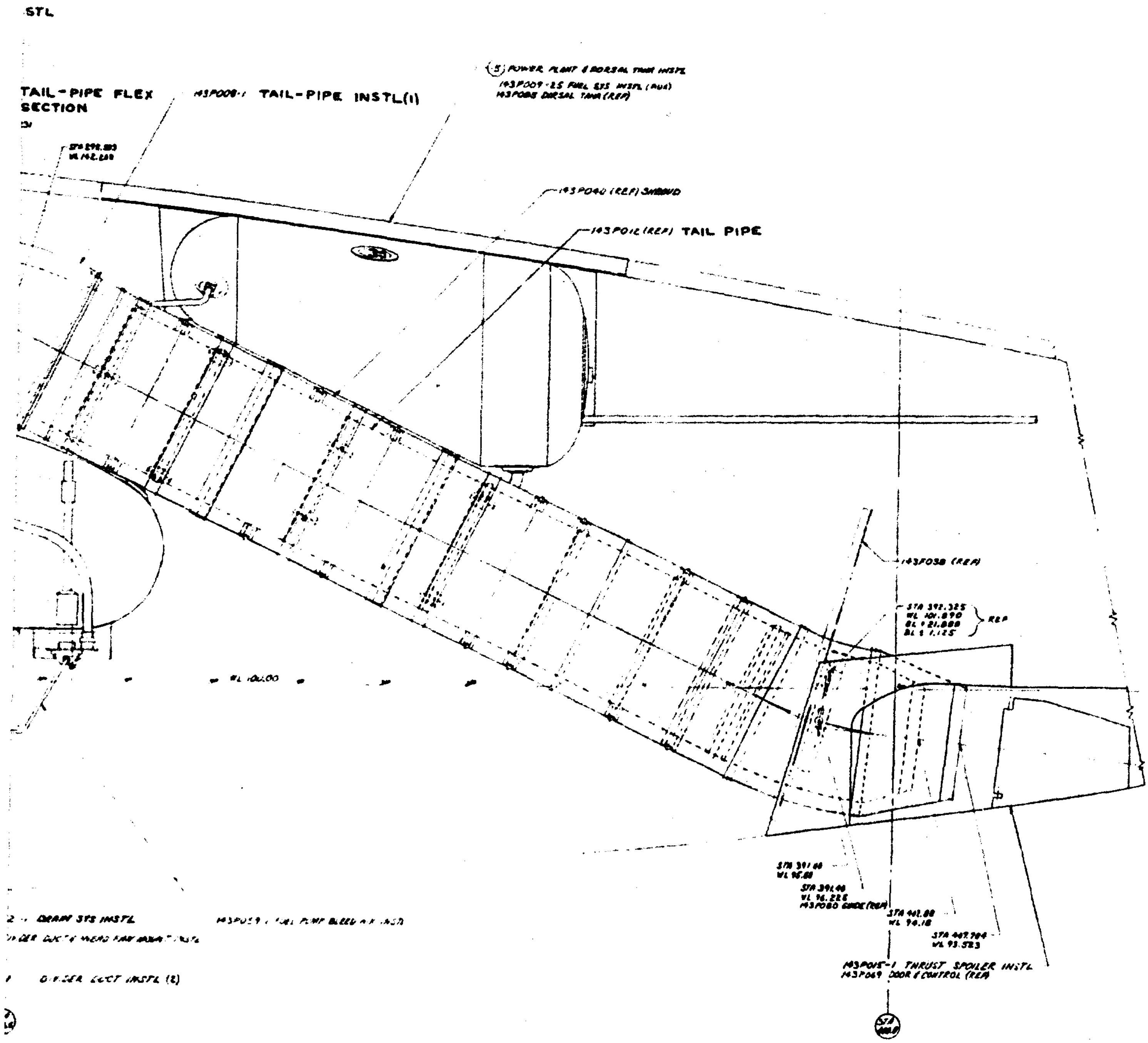
143P046-1 BULKHEAD INSTL
143P007-1 ACCESSORY INSTL
43P047 (REF)
143P042-1 STUB DUCT INSTL
143P033-1 BLEED DUCT INSTL
REMOVABLE
143P006-1 ENG AIR INLET INSTL
STA 215.38
BL 11.50
401202B-441 ENGINE (2)
LIFTING ATTACH (REF)
143P078-1 COOLING AIR DUCT INSTL
DRAINS —
SCD P0047-1 INSULATION INSTL-PF
143P029-1/-2 DUCTING ASSY PITCH FAN
143P026 (REF)
43P027 (REF)
143P028 (REF)
143P039-1 FUEL SYS INSTL
143P079-1 PANEL INSTL
143P059
143P061-1 DUCT INSTL
143P011-100 STARTING SYS INSTL

B



FIRE WALL INSTL
REMOVABLE
TAIL-PIPE FLEX SECTION
TAIL-PIPE INSTL(1)
143P032 (REF)
143P052 -1 DRAIN SYS INSTL
143P059 1 FUEL PUMP BLEED AIR INSTL
143P013 -1
DIVIDER DUCT INSTL (2)
SCDP00664 BLANKET INSULATION
POWER PLANT INSTL
SIDE VIEW
FWD
WL 100.00

C



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Nose-Fan Ducts

The nose-fan duct installations connect the divider ducts and nose-fan scrolls. Each duct installation utilizes two pin-jointed bellows in the forward angle section to accommodate the large thermal expansion of the duct and to eliminate external loads which would have required additional fuselage structural support. All static loads that would have been incurred by the use of free bellows are reacted by tension on the pin joints and duct walls. The swinging linkage that reacts the duct positive g-loads in tension during jet-mode maneuvers permits significant weight savings in the straight duct sections. The ducts are covered with foil-clad insulation blankets to reduce both hot gas thermal losses and area cooling requirements.

Nose-Fan Inlet Enclosure

The installation for the nose-fan inlet enclosure utilizes a fiber glass structure to provide the nose-fan bellmouth and inlet duct. Short-cord, contoured, foam-filled aluminum louvers are utilized to provide a smooth bellmouth closure during fan-mode operation and to minimize obstruction to the pilot's vision during fan-mode operation.

Nose-Fan Thrust Modulator Doors

The nose-fan thrust modulator is comprised of two titanium doors. They are contoured to the fuselage mold line on the outside and are contoured to act as a fan efflux turning vane on the inside. The doors pivot about longitudinal hinge lines. To maximize nose-fan vertical thrust reversal, each door carries a cascaded turning vane, and there is a longitudinal strut between the doors. Figures 2 and 38 show installation and configuration details.

Thrust-Spoiler Doors

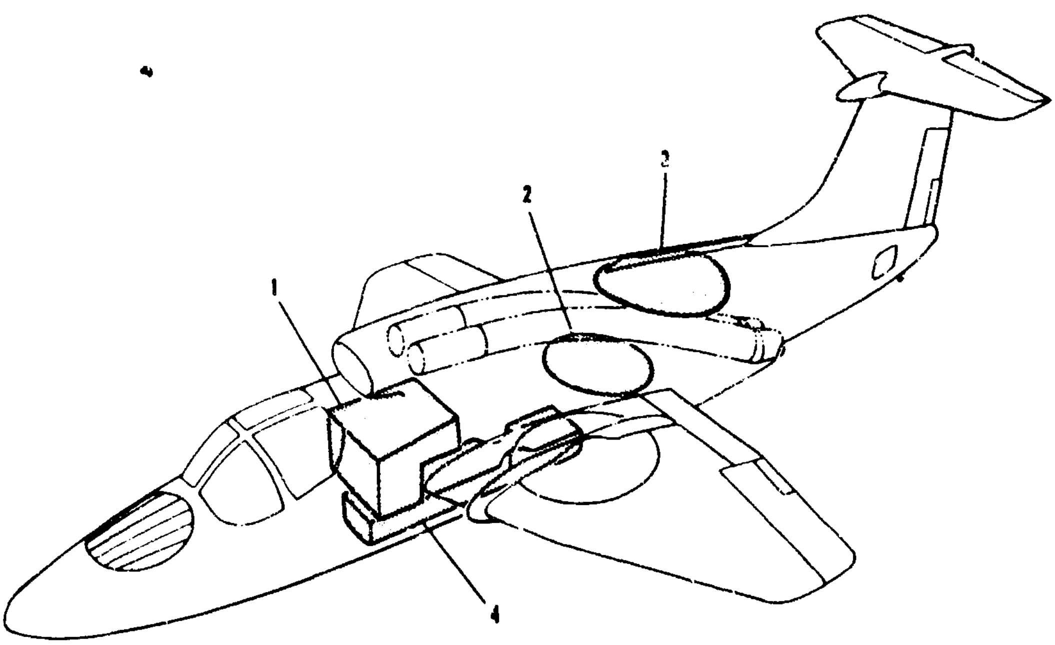
The thrust-spoiler-doors assembly, located aft of the tail-pipe nozzles, is provided to obtain a low forward thrust component at high engine-power levels. The spoilers were used only during taxiing; using them reduced the braking effort that would otherwise have been required to overcome the relatively high J-85 idle power residual thrust (see Figure 38).

Fuel System

The aircraft fuel system (see Figures 39 and 40) consists of a main fuel system and an auxiliary system which may be installed for extended range. The main system consists of a forward bladder cell having a 261-gallon (1749-pound) capacity, an aft aluminum tank having a 140-gallon (938-pound)

capacity, and a dorsal tank having a 117-gallon (784-pound) capacity. The dorsal tank is the second cell of the aft main tank. The extended-range auxiliary system consists of a lower internal fuselage tank having approximately a 125-gallon (837-pound) capacity. The total capacity with the auxiliary tank installed is 643 gallons (4308 pounds). The main fuel system permits each main tank to supply one engine and incorporates a cross-feed valve to permit either of the main tanks to supply both engines.

The engine pumps can draw fuel from the main tanks for up to 6000 feet without booster pumps. A booster pump is installed in each main tank to provide fuel for a distance over 6000 feet. The pumps are powered by engine compressor eighth-stage bleed air, which is controlled by normally open solenoid valves. All pumps are powered by either engine and by the No. 1 engine air starting duct. Each booster pump is capable of supplying both engines. The extended-range tank (which has never been installed) contains a transfer pump and valving to permit transfer of fuel to either of the main tanks. Each main tank and the extended-range tank contain a water sump with a drain valve. Capacitance-type fuel quantity gages indicate, in pounds, the amount of fuel remaining. Fuel strainers



1. Forward Main Fuel Tank
2. Aft Main Fuel Tank
3. Extended Range Dorsal Tank
4. Extended Range Belly Tank

Figure 39. Fuel System Tank Location Diagram.

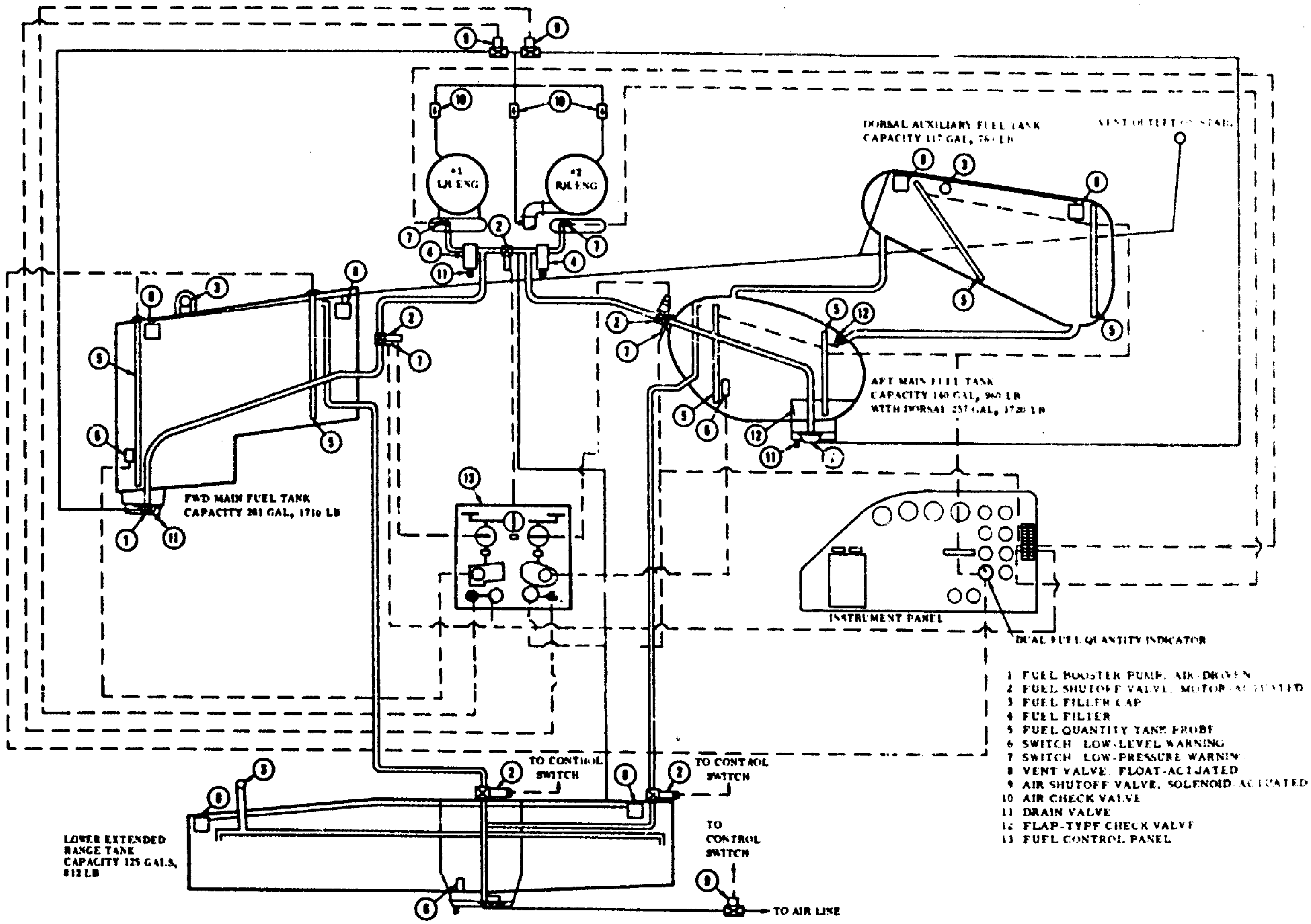


Figure 40. Fuel System Schematic.

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are provided upstream of each engine. Float-switch-operated warning lights indicate low fuel level, and pressure-switch-operated warning lights indicate low fuel pressure and inoperative booster pump.

Engine Fire Warning System

An independent overheat and fire detection indicating system is installed for each engine. Engine compartment overheating is indicated by a flashing red light in the fire warning system on the cockpit main instrument board. Fire is indicated by a steady red glow.

Engine Fire Extinguishing System

The fire extinguishing system (see Figure 41) consists of two twin-valve pressure vessels, which use a chemical agent that is pressurized to 600 psi and that can be discharged at a high rate into the forward and aft engine compartments of either engine selected.

Accessory Installations

Each accessory installation consists of a drive shaft, two cooling blowers, a gearbox, a 28-volt dc generator, and a 3000-psi hydraulic pump. Each J-85 engine drives one accessory installation from its power takeoff pad. The drive shaft couples the gearbox to the J-85 power takeoff pad. The gearbox contains a straight-through shaft, an idler shaft, and a cross shaft. The cooling blowers are mounted on the gearbox and are driven by the cross shaft. The generator is mounted on the gearbox and is driven by the straight-through shaft. The hydraulic pump is mounted on the back side of the generator and is driven directly by the generator shaft.

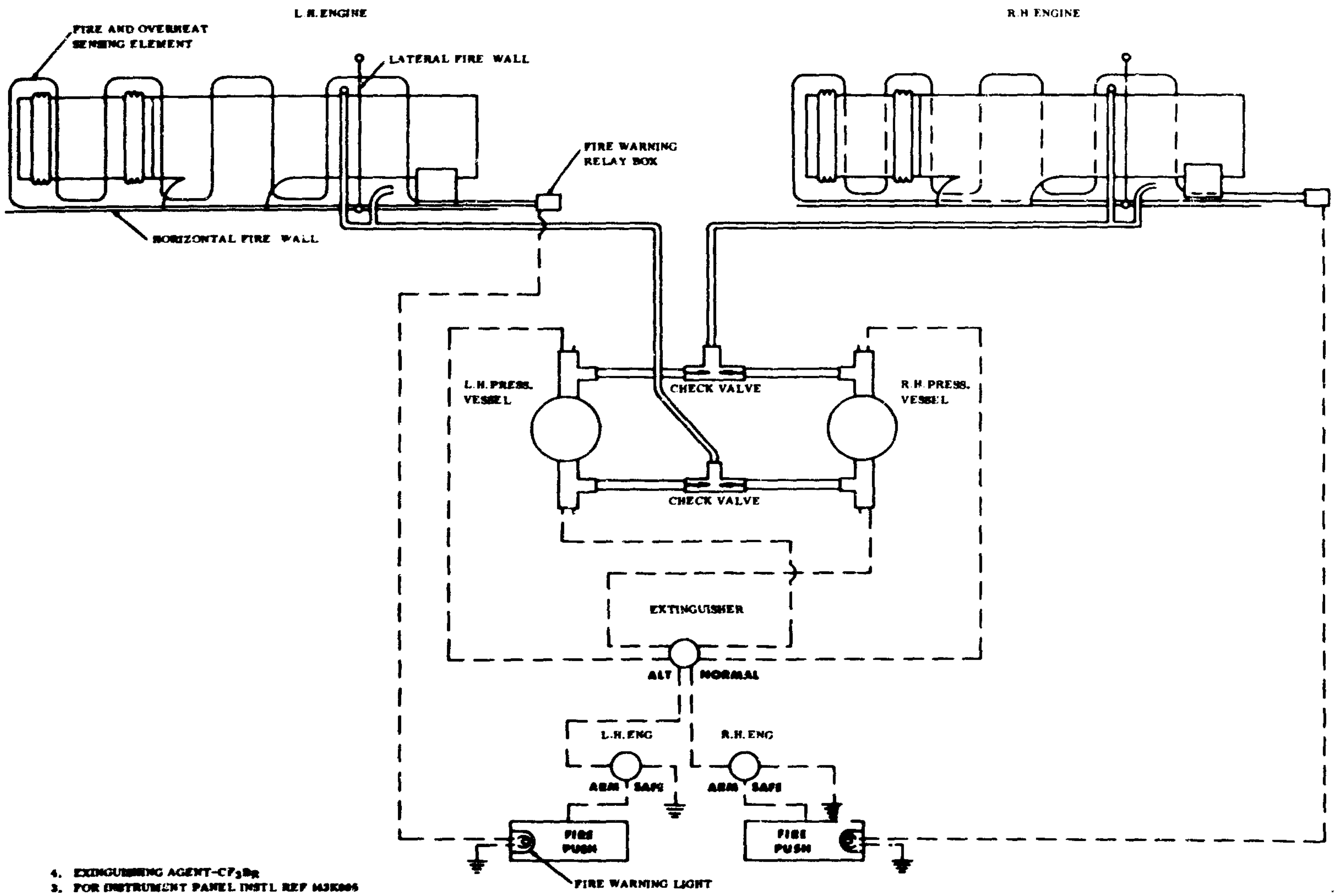
Air Impingement Starter System

An air impingement starter system is provided. Starter ducts are routed from each engine and terminate at an external connector installation located in the lower portion of the center fuselage bay. System installation provides for individual engine starting. Check valves are provided in each engine starter duct to prevent reverse (compressor bleed) flow.

MAINTAINABILITY HISTORY AND RECOMMENDATIONS

Failure Rate

Forty-three propulsion system installation failures were reported. Of these, five were reported during a flight or ground test. The remaining



4. EXTINGUISHING AGENT–CF_3Br
3. FOR INSTRUMENT PANEL INSTL REF 143K005
2. FOR ELECTRICAL INSTL REF 143E018
1. FOR PLUMBING REF 143P017

Figure 41. Fire Warning and Extinguishing System.

38 failures were reported during the various inspection and functional test procedures. The value of effective preflight maintenance in preventing inflight failures is clearly indicated. The man-hours expended in correcting discrepancies are shown in Table VIII.

The highest failure rate for the propulsion system installation components was in the thrust-spoiler-doors assembly. The thrust spoilers were not flight evaluated to verify the expected contribution to aircraft handling qualities for jet-to-fan conversions. This was attributed primarily to lack of confidence in spoiler reliability.

The component that required the most maintenance man-hours for the total number of failure repairs reported is the tail-pipe flex section.

The propulsion system elements itemized below may be satisfactory as they are now, or they may require only some slight configuration modification or procedural change to make them satisfactory. Little or no effort beyond identification of these potential problem areas has been possible because of the limited nature of this study. Because of the existing uncertainties, further study of each element is recommended. The study should establish basic criteria and should include a preliminary evaluation of critical factors to determine the necessity for, and objectives of, either appropriate corrective action or more detailed study.

1. Engine and fan installations - accessibility/maintainability.

2. Gearbox fan-assembly installation - accessibility/maintainability.

3. Fuel distribution subsystem, including the following:

    a. Booster pump capacity.

    b. Check valves.

    c. Fuel control valves.

    d. Fuel filters (low-pressure, line-mounted versus high-pressure, engine-mounted).

    e. Fuel management panel (configuration, location, and methods of operation).

    f. Fuel transfer provisions and procedures.

    g. Plumbing details.

TABLE VIII. MAN-HOURS EXPENDED IN CORRECTING DISCREPANCIES IN PROPULSION SUBSYSTEMS

Part Name	Prepa-ration	Diag-nosis	Accessi-bility	Corrective Action	Reas-sembly	Check-out	Total Man-Hours	Remarks
Fire Extinguisher Bottles	0.1	0	0.5	3.0	0.5	0	4.1	Bottles recharged by vendor
R.H. J-85 Engine Mounts	1.0/0.3*	0.2	4.0/1.5	24.0/10.0	18.0	2.5	49.7	Mounts reworked - engine removed
R.H. J-85 Divider Duct Liner	0.5	0.1	12.0/4.0	2.0	12.0/4.0	0.1	26.7	Liner flared - 1 engine removed
R.H. Engine Bellows Clamp	0.1	0.1	12.0/4.0	0.5/0.3	12.0/4.0	0.1	24.8	Clamp replaced - 1 engine removed
L.H. J-85 Engine Mounts	1.0/0.3	1.0	4.0/1.5	30.0/15.0	18.0/6.0	2.5/0.5	56.5	Mounts reworked - engine removed
Wing-Fan Bellmouth	0.1	0.2	0	0.3	0	0	0.6	Crack stop-drilled
Pitch-Fan Inlet Louver Mounting	0.1	0.1	0.3	1.0	0.4	0.1	2.0	New bushing installed
Fuel System (Cross-Feed) Ball Valve	0.1	0.1	0.2	1.2	0.2	0.2	2.0	Replaced
Engine Inlet Assembly	0.2	0.1	0.1	0.5	0.1	0.5	1.5	Chafing point cleared
Exhaust Deflector Strake	0.2	0.1	0.5	1.2/0.5	0.5	0	2.5	Removed and welded
L.H. Wing-Fan Scroll Lever Arms	0.2	3.0	64.0/16.0	2.0	88.0/22.0	10.0/1.0	167.2	Welded and machined
Fuel Tank Assembly - Aft	0.1	3.0/1.5	3.0/1.5	3.0	2.0/1.0	1.0	12.1	Leak repaired
Divider Duct Ring Section	0.1	0.2	0	0.5	0	0.1	0.9	Reformed to print
Divider Duct Stamping	0.1	0.1	0.1	0.5	0.1	0	0.9	Crack repaired
Shutoff Valve, Bleed Air Fuel Booster Pump	0.3	0.2	0.2	0.7	0.3	0.2	1.9	Replaced; stalled when 50 psi or greater was applied
Nut Plate, Tail-Pipe Assembly	0.2	0.1	2.0/1.0	2.0/1.0	3.0/1.5	0.8	8.1	Reassembled and used (locking device had loosened)
Strake	0.2	0	1.0	0.5	1.3	0	3.0	Crack welded
R.H. Flexible Section Tail Pipe	2.0	0	1.0	1.0	2.0	0.1	6.1	Cracks in the seam weld welded
L.H. Flexible Section Tail Pipe	3.0	0.1	1.0	1.0	2.0	0.1	7.2	Cracks between strap and bushing welded
Bellmouth, Inlet Installation Pitch Fan	0.3	0	1.0	0.7	1.0	0	3.0	Separation between bellmouth and fiber glass hot section rebounded
Valve Check, J-85 Engine Starting System	0.2	0.1	0.5	0.5	0.5	0.2	2.0	One of the springs on one of the flapper valves replaced
Line Assembly, J-85 Engine Starting System	0.2	0.5	0.5	10.0/5.0	0.5	0.3	12.0	Line ruptured by overheating repaired
Valve Check, J-85 Engine Starting System	0.2	1.0	0.5	3.5	0.5	0.3	6.0	Sharp edges on flapper valve that caught on valve body repaired
L.H. Thrust Spoiler Door	0.5	0.1	0	0.3	0.2	0.2	1.3	Crack welded
R.H. Thrust Spoiler Door	0.5	0.1	0	0.2	0.2	0.2	1.2	Crack welded
L.H. Thrust Spoiler Door	0.5	0.1	0	1.0	0.2	0.2	2.0	Crack welded
R.H. Thrust Spoiler Door	0.1	0.1	0	1.0	0	0.2	1.4	Crack welded
Thrust Spoiler Bracket	0.1	0.1	0.1	0.5	0.1	0.1	1.0	Welded
Thrust Spoiler Channel	0.1	0.1	0.2	2.5/1.2	0.4/0.3	0	3.3	Crack stop-drilled

*If 2 numbers are divided by a slash, the first number represents the man-hours; the second, the clock hours.

Best Available Copy

4. Fire extinguishing subsystem.

5. Seals in general, but particularly the following:

   a. Divider duct - horizontal fire wall.

   b. Tail-pipe fairing.

   c. Fan wing.

6. J-85 lateral fire wall (compressor-combustion section separation).

7. Joint at upper tail pipe and shroud.

8. Propulsion system installation elements associated with engine compressor stall suppression.

DESIRABLE FEATURE

In fan-mode operation, the divided hot gas ducting eliminates the asymmetric forces that would result from a single engine's being out.

UNDESIRABLE FEATURES

1. Thrust-spoiler-doors installation.

2. Tail-pipe flex section.

3. Excessive man-hours required for engine and fan installations.

4. Excessive man-hours required for gearbox fan-assembly installation.

RECOMMENDATIONS AND SUGGESTIONS FOR UNEVALUATED IMPROVEMENTS

Elimination of Thrust Spoilers

The elimination of thrust spoilers on future aircraft should be considered. The present thrust spoilers are cracking and thus present a problem. If required on a unit in the future, they will have to be built of heavier material and have stronger hinge points. The actuation assembly should be

designed to have no overcenter travel on the control rods. If the spoilers are included on new aircraft, they should be designed for easy removal, since they must be removed when the tail pipes are removed.

Accessibility to Pitch Controller Door Actuator

Easier access should be provided to the pitch controller door actuator and controls. This could be accomplished partly by installing the nut plates or fasteners so that they can be easily reached. At present, the screws are hidden and are covered by a sealant. Also, installation of heat shield blankets in this area is very time-consuming.

Accessibility to Engines

Access to the engines should be improved. This could be accomplished by use of a two-piece engine cowl with quick-type fasteners for installation instead of the one-piece design with machine screws for installation. At present, the total cowl assembly must be removed to gain access to just one engine.

Installation of Sheet Metal Clamps

The sheet metal clamps are installed on flanges. This is a problem, since the outside diameters of the clamps are of different sizes. In addition, the sheet metal clamps do not hold their shape in this type of installation because of the difference in the diameters of the flanges.

Access Door in Forward Engine Mount Structure

An access door should be installed in the right-hand side of the forward engine mount structure. To provide for direct vision into the throttle assembly area, the access door should be removable.

Redesign of Micro Adjustments

Flexible-type cable should be installed in the aircraft at the time of manufacture so that the area surrounding the cable can be maintained without removing the cable and tubing. With the cable installed, micro adjustments can be made that will make it easy to set the throttles together in the cockpit and to trim both engines to the same rpm. The rpm trim has been a real problem each time an engine has been installed.

Redesign of Insulation

All insulation on pitch-fan ducting, bellows, and crossover ducts should be redesigned for easier installation. The insulation has to be removed

to inspect the bellows and pipe. Insulation removal is time-consuming, since spot tacking is required to seal the present insulation.

Installation of One-Piece Fire Wall

A one-piece vertical fire wall instead of sectionalized panels should be provided between engines.

Installation of Larger Bleed Valve Opening

For accessibility, a larger bleed valve opening should be provided in the engine forward support cowl.

Improvement of Pitch-Fan Louver Actuator Links

The pitch-fan louver actuator links should be improved both in their assembly and in the strength of the links and rod ends.

Improvements in Trim Adjustment

Fan scroll adjustment access for EGT trim adjustments on wing fans should be improved. The engine tail-pipe nozzle should be redesigned so that rats (metal plates) can be used for ease of trimming.

FLIGHT CONTROLS SYSTEM

PRIMARY FLIGHT CONTROLS SYSTEM CONFIGURATION AND OPERATION

General Description

The primary flight controls system is made up of the following elements:

1. Jet mode:

   a. Lateral (ailerons).

   b. Directional (rudder).

   c. Longitudinal (elevators).

   d. Thrust (throttles, J-85 power).

2. Fan mode (see Figure 42):

   a. Lateral (wing-fan louvers, differential collective stagger, wing to wing).

   b. Directional (wing-fan louvers, differential vector, wing to wing).

   c. Longitudinal (nose-fan thrust modulation).

   d. Lift (wing-fan louvers and nose-fan thrust modulator doors, collective stagger).

   e. Thrust vector (wing-fan louvers, collective vector).

   f. Lift/thrust (throttles, J-85 power).

3. Conversion:

   a. Wing-fan inlets (wing-fan doors).

   b. Longitudinal trim (horizontal stabilizer, automatic programmed position, jet or fan mode).

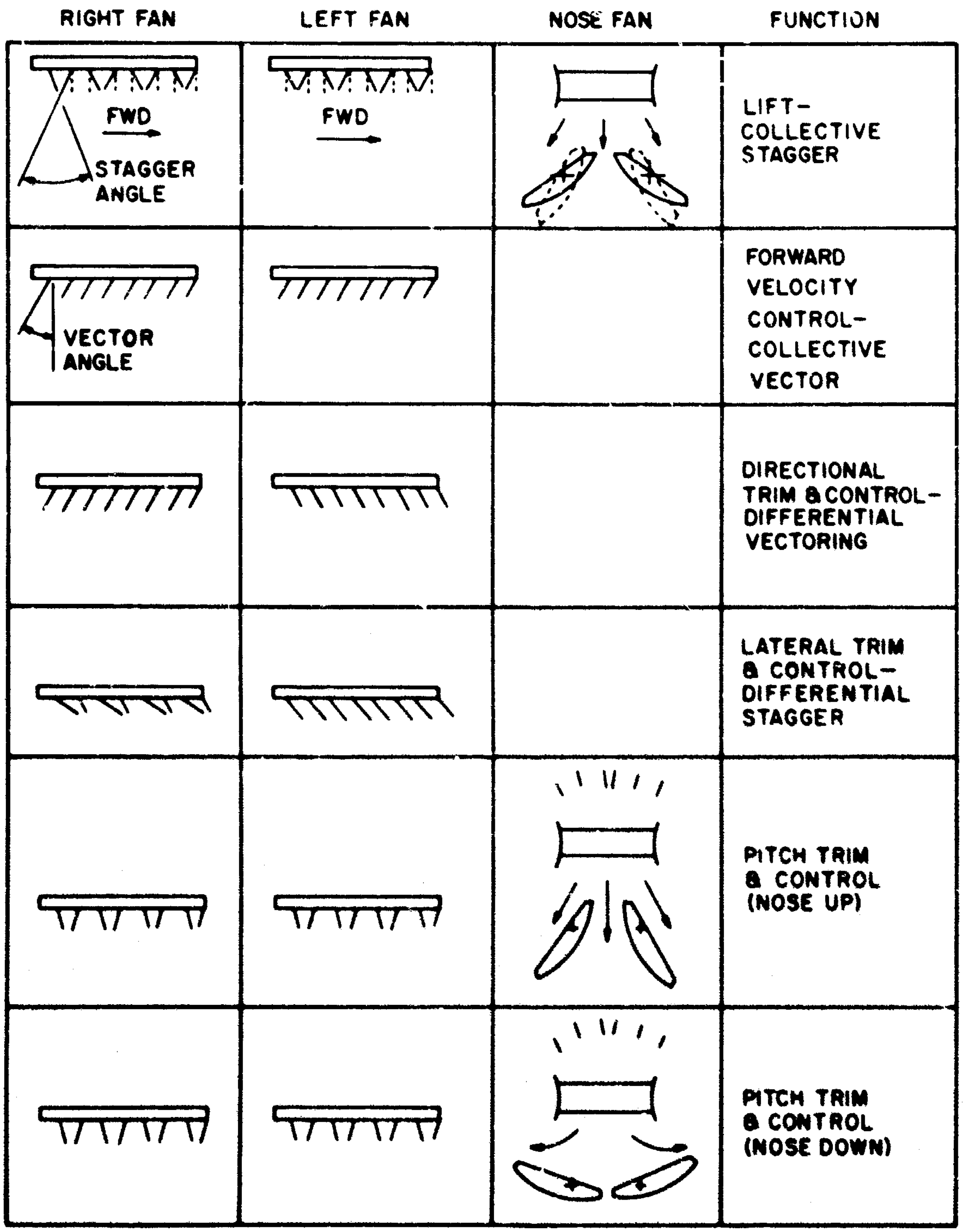


Figure 42. Fan-Mode Flight Control System Operation.

c. Flight mode (diverter valves, J-85 power distribution, jet or fan mode).

4. Conversion control interlock system.

The primary flight control system consists of a conventional stick, rudder pedals, and a collective lift stick mechanically connected to aerodynamic flap-type control surfaces, wing-fan exit louver servo actuator valves, and a nose-fan thrust modulator servo actuator valve. Ailerons, wing-fan exit louvers, and nose-fan thrust modulator doors are hydraulically actuated. Actuation of wing-fan louvers is accomplished by using two actuators per fan (one forward, one aft) to perform both vector and stagger functions; an even-odd louver actuation scheme is used. Nose-fan thrust modulation is accomplished with one actuator. Wing-fan exit louver and nose-fan thrust modulator actuator servo valves have electrical input features capable of accepting actuator position input signals from the stability augmentation system (SAS) amplifier. The servo systems incorporate tandem-type actuators powered by separate engine-driven hydraulic systems.

Stick and rudder pedals perform identical attitude control functions in jet-mode and fan-mode operations. The collective lift control provides for altitude control in the fan-flight mode by adjusting the wing-fan exit louver and the nose-fan thrust modulator door stagger. Lateral stick motion controls the ailerons and the differential stagger of wing-fan exit louvers. Longitudinal stick motion controls elevators and nose-fan thrust modulators. Rudder pedals control the rudder and differential vector of the wing-fan exit louvers and the wheel brakes. The vector switch on the conventional stick controls the exit louver collective vector angle and provides the fan-mode forward velocity control. A mechanical mixer is installed between the cockpit controls and the louver actuator servo valves. This mixer interprets pilot commands and positions the wing-fan exit louvers. An electrical mixer controls the nose-fan thrust modulator. The mechanical mixer provides automatic "washout" (that is, decreasing and eventual decoupling) of wing-fan exit louver response to pilot commands as a function of louver vector angle (forward speed), thereby inactivating the wing-fan control system in the conventional-flight mode while always retaining full aerodynamic control authority. The pitch portion of the mechanical mixer decouples the nose-fan thrust modulator doors.

The aircraft is designed to be controllable in jet-mode flight if both hydraulic systems are inoperative and to be controllable in fan-mode flight if the SAS is inoperative. The flight controls, their functions, and their locations are shown in Figure 43.

Lateral System

The lateral (roll) system utilizes ailerons and differential stagger (Figure 43) of the wing-fan exit louver system. The aileron is actuated by a hydraulic/aerodynamic servo system. Differential stagger is controlled by a hydraulic servo system. The lateral control system is actuated by the pilot through the use of push rods and bell cranks. Pilot commands pass through the mechanical mixer and transmit a control signal to the forward and aft louver actuators. In jet-mode flight, pilot control force gradient and stick centering forces are obtained from adjustable geared trailing-edge tabs. This arrangement also provides aerodynamic servo-controlled ailerons for backup lateral control during jet-mode flight if both hydraulic systems become inoperative. The tabs are mass balanced. In fan-mode flight, artificial force feel with stick centering is provided. Fan-mode control is decoupled by the mechanical mixer during jet-mode flight.

Directional System

The directional (yaw) system utilizes a rudder and differential vector (see Figure 43) of the wing-fan exit louver system. The rudder is actuated by using a push-rod cable system with a tension-regulating cable drum. Differential vector is controlled by the same (roll) hydraulic servo system. The system is mechanically actuated by the pilot through the use of push rods and bell cranks. Pilot commands pass through the mechanical mixer and transmit a control signal to the forward and aft louver actuators. The rudder is mass balanced. In jet-mode flight, aerodynamic forces on the rudder provide control force gradient and control centering. In fan-mode flight, an artificial rudder force gradient is provided. In jet-mode flight, fan-mode control is decoupled by the mechanical mixer.

Longitudinal System

The longitudinal (pitch) system utilizes elevators and nose-fan thrust modulator doors (see Figure 43). Elevator actuation bypasses both mechanical and electrical mixers and is accomplished by using a push-rod and cable system with a tension-regulating cable drum. The nose-fan thrust modulator doors are controlled by a hydraulic servo system. The servo system is mechanically actuated by the pilot through the pitch portion of the mechanical mixer by a push-rod and bell-crank system. The pitch input provides summed pitch (attitude) and collective stagger (altitude) output commands; it also provides automatic transition pitch trim as a function of mechanical mixer collective vector commands, which are also transmitted to the pitch mixer by a push-rod and bell-crank system. The elevators are mass balanced. In jet-mode flight, aerodynamic forces on the elevators provide the control force gradient

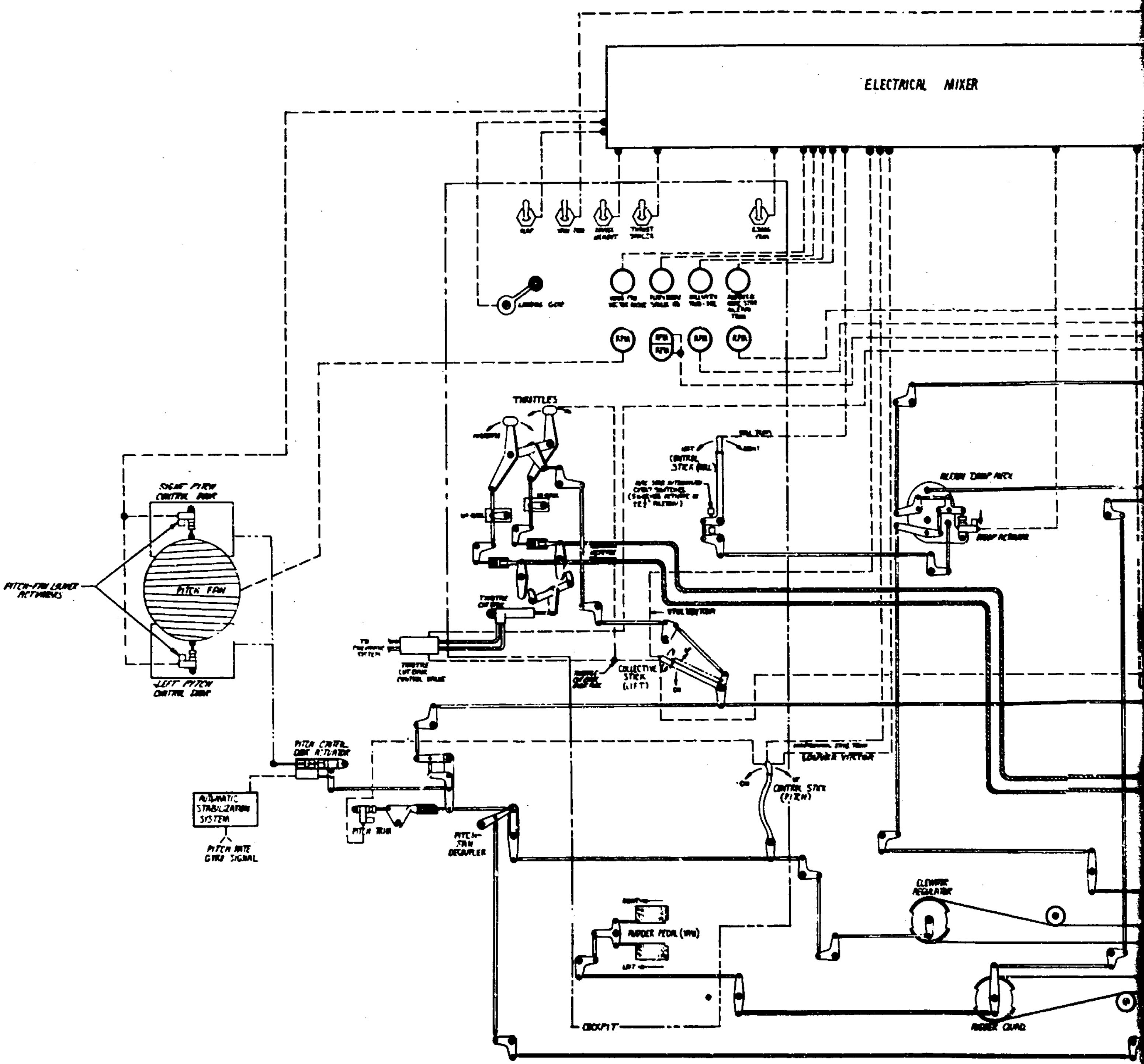
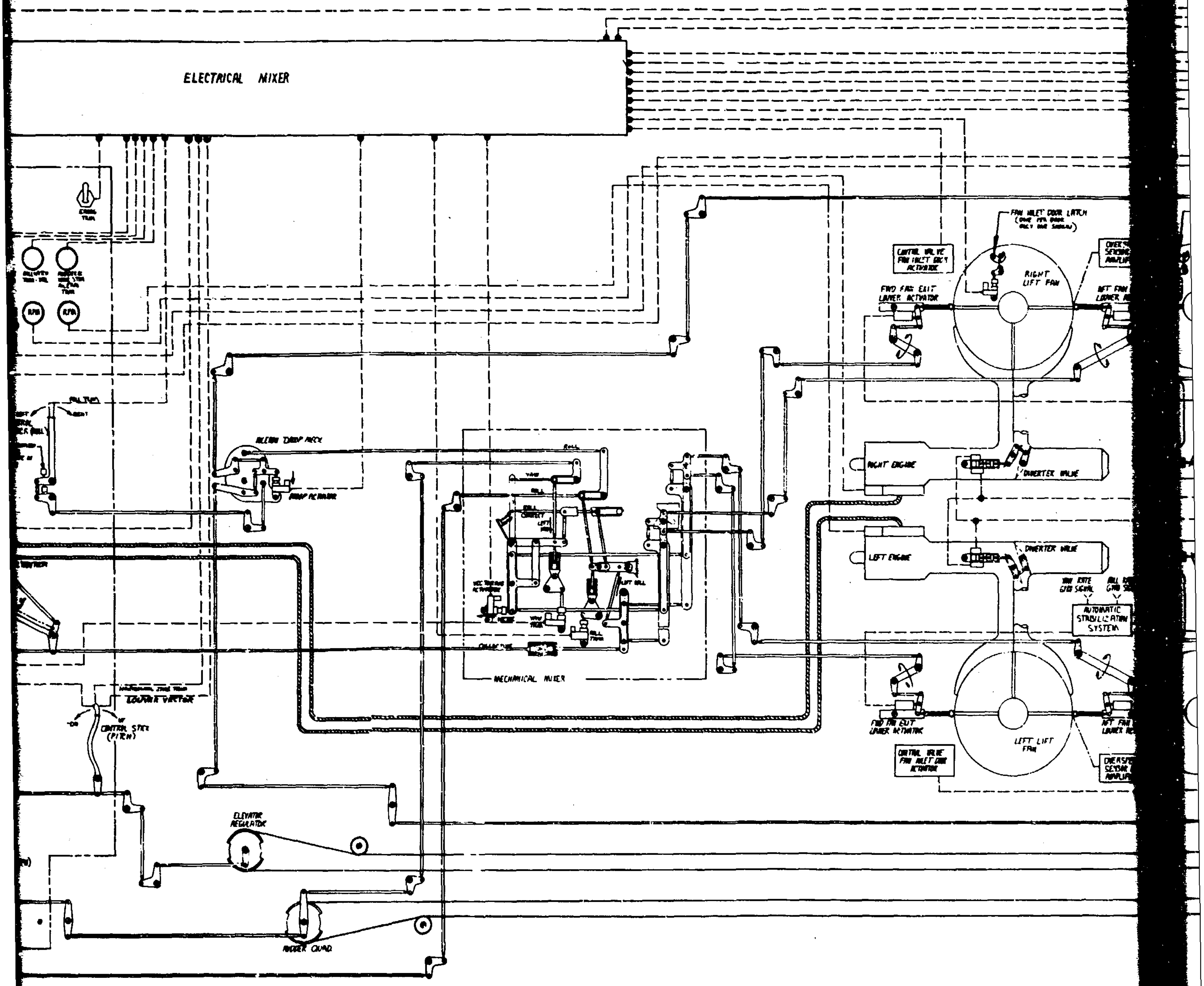
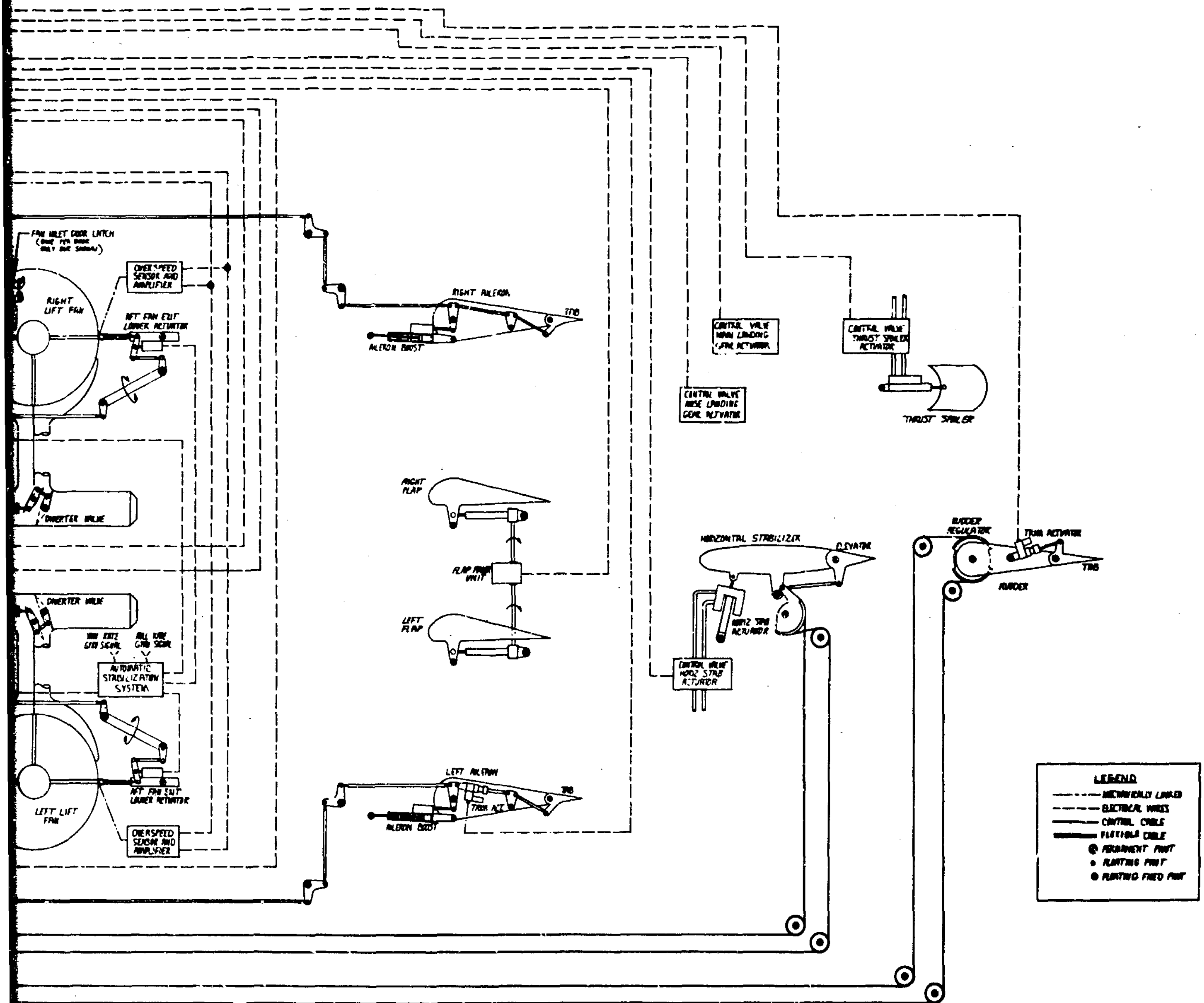


Figure 43. Flight Controls Schematic.


ELECTRICAL MIXER
FWD FAN EXIT LOUVER ACTUATOR
RIGHT LIFT FAN
RIGHT ENGINE
DIVERTER VALVE
LEFT ENGINE
DIVERTER VALVE
AUTOMATIC STABILIZATION SYSTEM
LEFT LIFT FAN
FWD FAN EXIT LOUVER ACTUATOR
AILERON DROOP NECK
ROLL
MECHANICAL MIXER
COLLECTIVE
LOUVER VECTOR
CONTROL STICK (PITCH)
ELEVATOR REGULATOR
RUDDER QUAD

B



FAN INLET DOOR LATCH
OVERSPEED SENSOR AND AMPLIFIER
RIGHT LIFT FAN
AFT FAN EXIT LOUVER ACTUATOR
DIVERTER VALVE
DIVERTER VALVE
YAW RATE GYRO SIGNAL
ROLL RATE GYRO SIGNAL
AUTOMATIC STABILIZATION SYSTEM
AFT FAN EXIT LOUVER ACTUATOR
LEFT LIFT FAN
OVERSPEED SENSOR AND AMPLIFIER
RIGHT AILERON
TAB
AILERON BOOST
RIGHT FLAP
FLAP DRIVE UNIT
LEFT FLAP
LEFT AILERON
TAB
TRIM ACT
AILERON BOOST
CONTROL VALVE MAIN LANDING GEAR ACTUATOR
CONTROL VALVE NOSE LANDING GEAR ACTUATOR
CONTROL VALVE THRUST SPOILER ACTUATOR
THRUST SPOILER
HORIZONTAL STABILIZER
ELEVATOR
HORIZ STAB ACTUATOR
CONTROL VALVE HORIZ STAB ACTUATOR
RUDDER REGULATOR
TRIM ACTUATOR
TAB
RUDDER
LEGEND
MECHANICALLY LINKED
ELECTRICAL WIRES
CONTROL CABLE
FLEXIBLE CABLE
PERMANENT PIVOT
FLOATING PIVOT
FLOATING FIXED PIVOT

C

and control centering. In fan-mode operation, artificial control force gradient and control centering are provided. During all operations at and above 45° collective stagger, the fan-mode control surfaces are mechanically decoupled from the pilot's controls by the mixers.

Lift System

The lift (altitude) system utilizes collective stagger (see Figure 43) of the wing-fan exit louver system and the nose-fan thrust modulator doors. Control is accomplished by a hydraulic servo system. The system is mechanically actuated by the pilot through the use of push rods and bell cranks connected to servo valves. Pilot commands pass through the mechanical mixing mechanisms and transmit control signals to the wing-fan louver actuators and the nose-fan modulator actuator. In jet-mode flight, the lift-control system is decoupled by the mixers.

Cockpit Throttles

Conventional cockpit throttles are provided at the left of the pilot; they provide individual or simultaneous control of the engines. A twist grip on the collective lift stick provides simultaneous control of both engines when the throttles are locked together.

Conversion Control Interlocking System (CCIS)

The CCIS provides for controlling, operating, and sequencing, and for interlocking transition, conversion, and preconversion functions. It contains both primary and standby circuits that are energized at the same time; however, only the selected circuit is in control at any one time. Included is the provision for pilot option to abort a conversion at any time during the sequence and to return the aircraft to the previous mode configuration. The transition (fan-mode operation) functions controlled by the CCIS are the horizontal stabilizer position, fan and jet trim, fan overspeed cutback authority, fan rpm indicators, and the SAS electrohydraulic servo actuator electrical outputs. The conversion functions controlled by the CCIS are shown in Figures 44, 45, 46, and 47. The preconversion functions controlled by the CCIS are the wing-fan door latches, the vectoring control (vectored from 45° to closed position of louvers), and the pitch-fan inlet louvers.

The basic conversion functions consist of sequential operation of the wing-fan doors, the horizontal stabilizer, and the diverter valves. However, certain other configuration prerequisites must be met before a conversion can be accomplished. These prerequisites depend on the direction of conversion (that is, jet to fan or fan to jet); the relationships are shown in Figures 44 and 45.

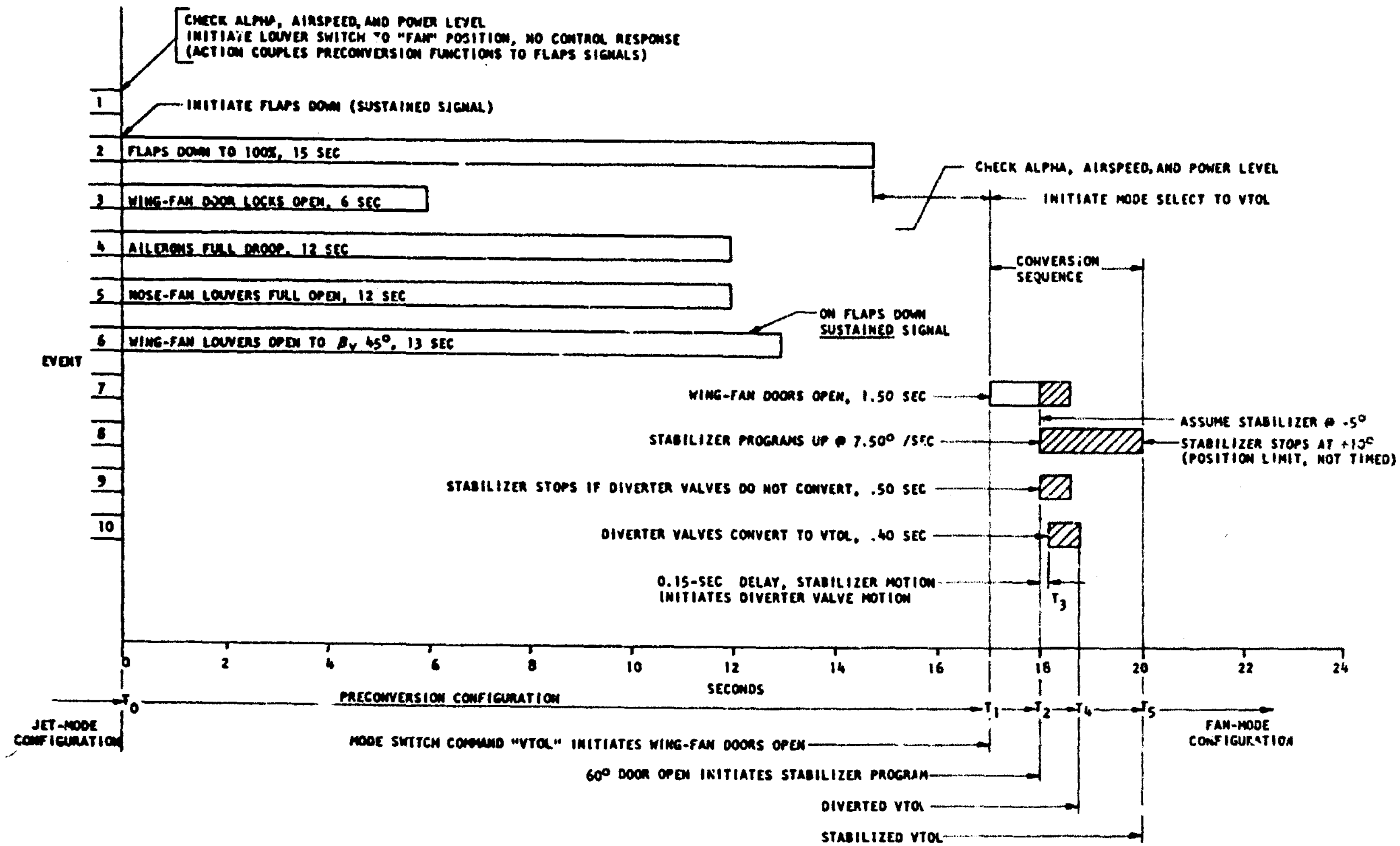


Figure 44. Conversion Events - Jet Mode to Fan Mode.

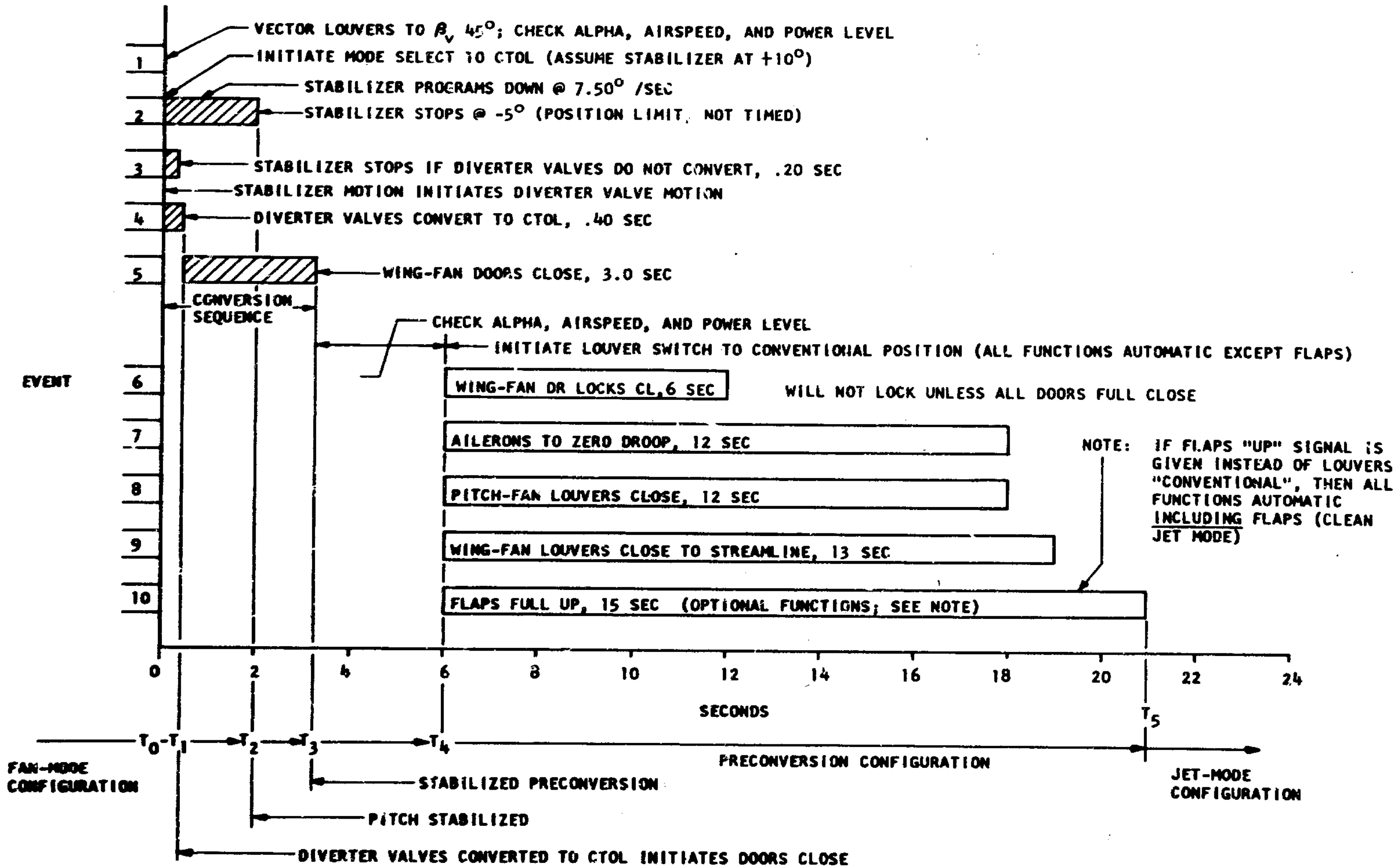
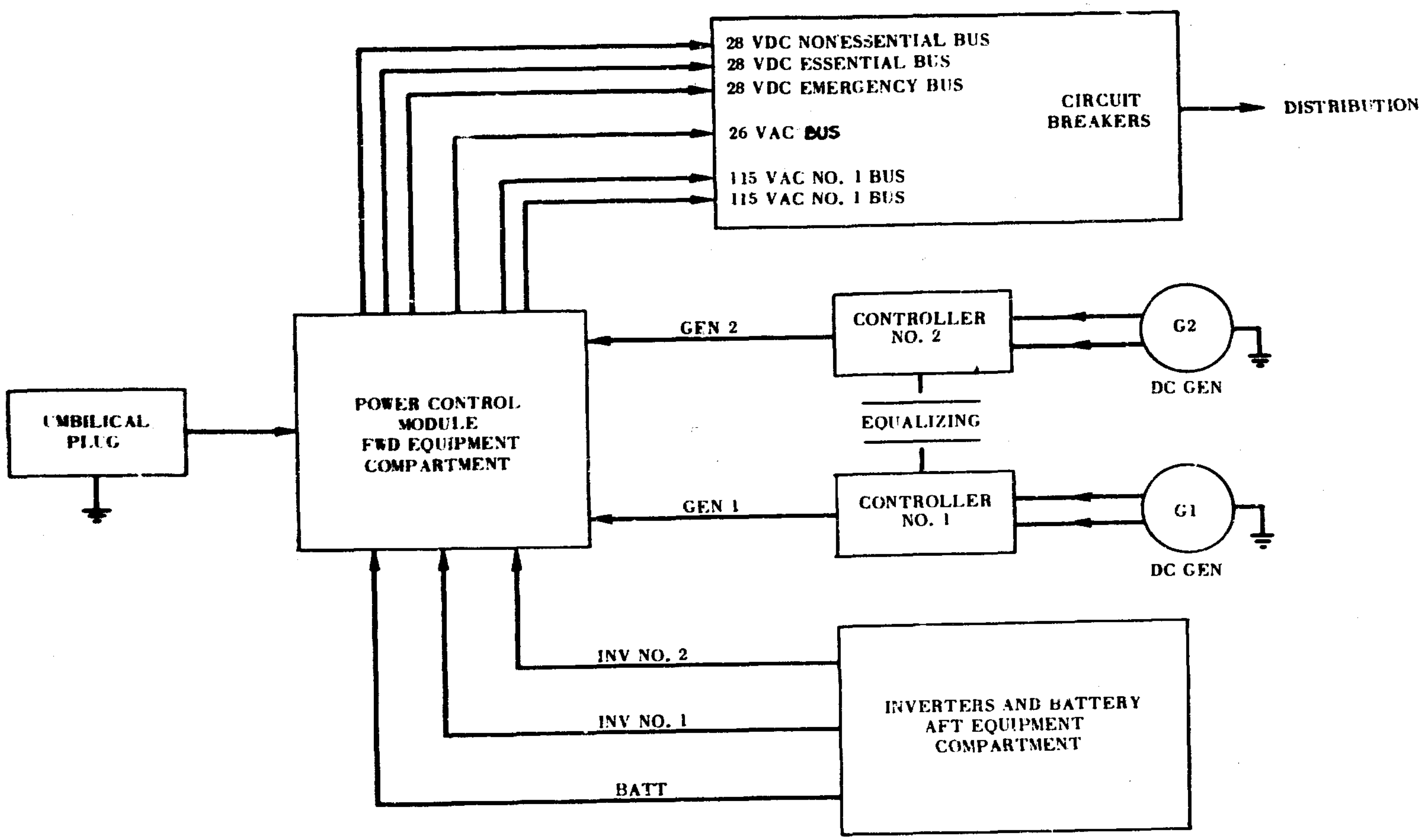


Figure 45. Conversion Events - Fan Mode to Jet Mode.



GENS:	28 VDC/165 AMP/RECTIFIED ALTERNATORS
INV'RS:	115 V 400 CPS SINGLE PHASE/250 VA
BATTERY:	28 VDC/25 AMP HR/SILVER ZINC 18 LBS
EMERGENCY:	60 AMP FOR 5 MIN/23.5 V CUTOFF

Figure 46. Conversion Control Interlock for Jet-Mode to Fan-Mode Conversion.

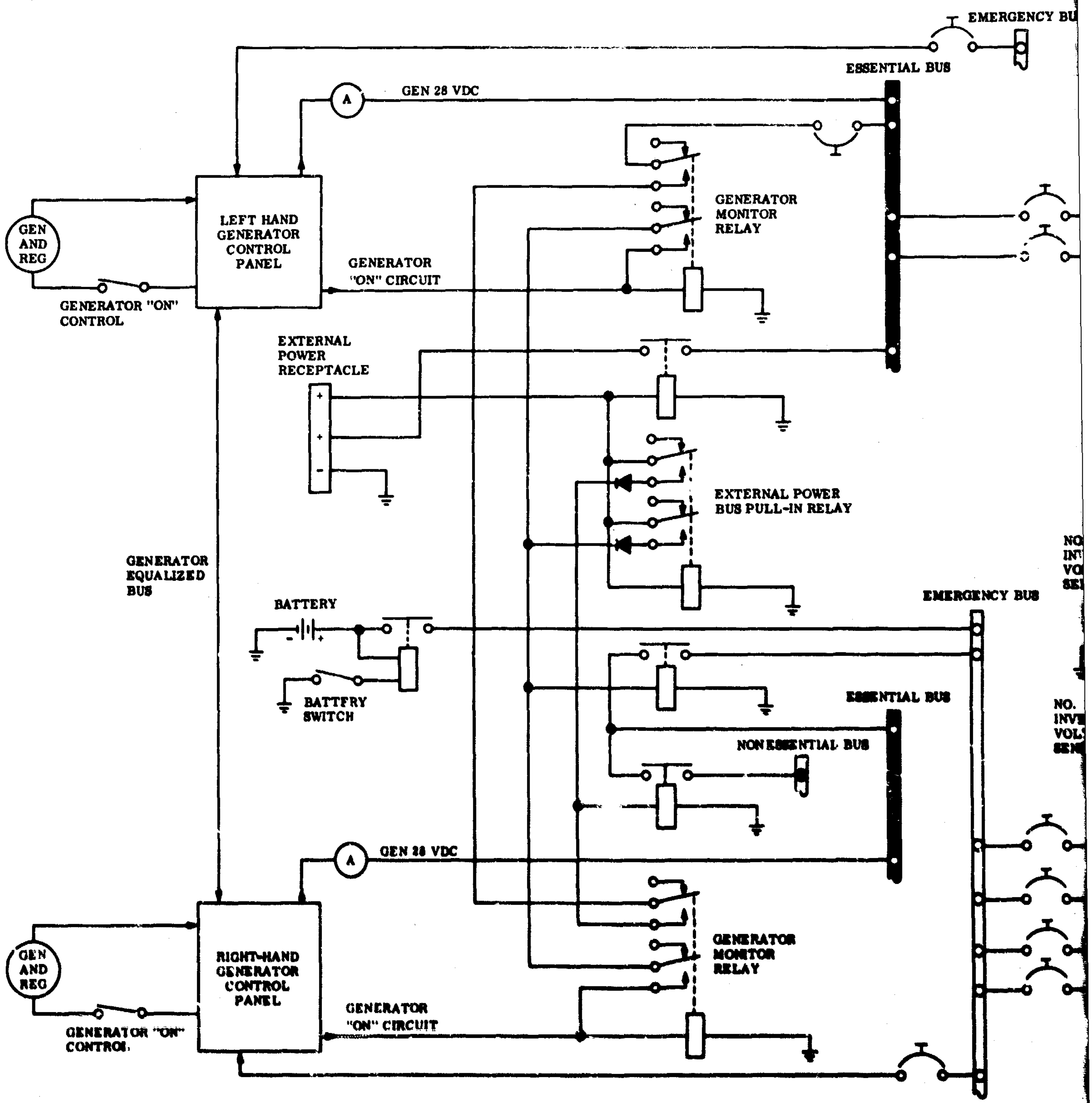
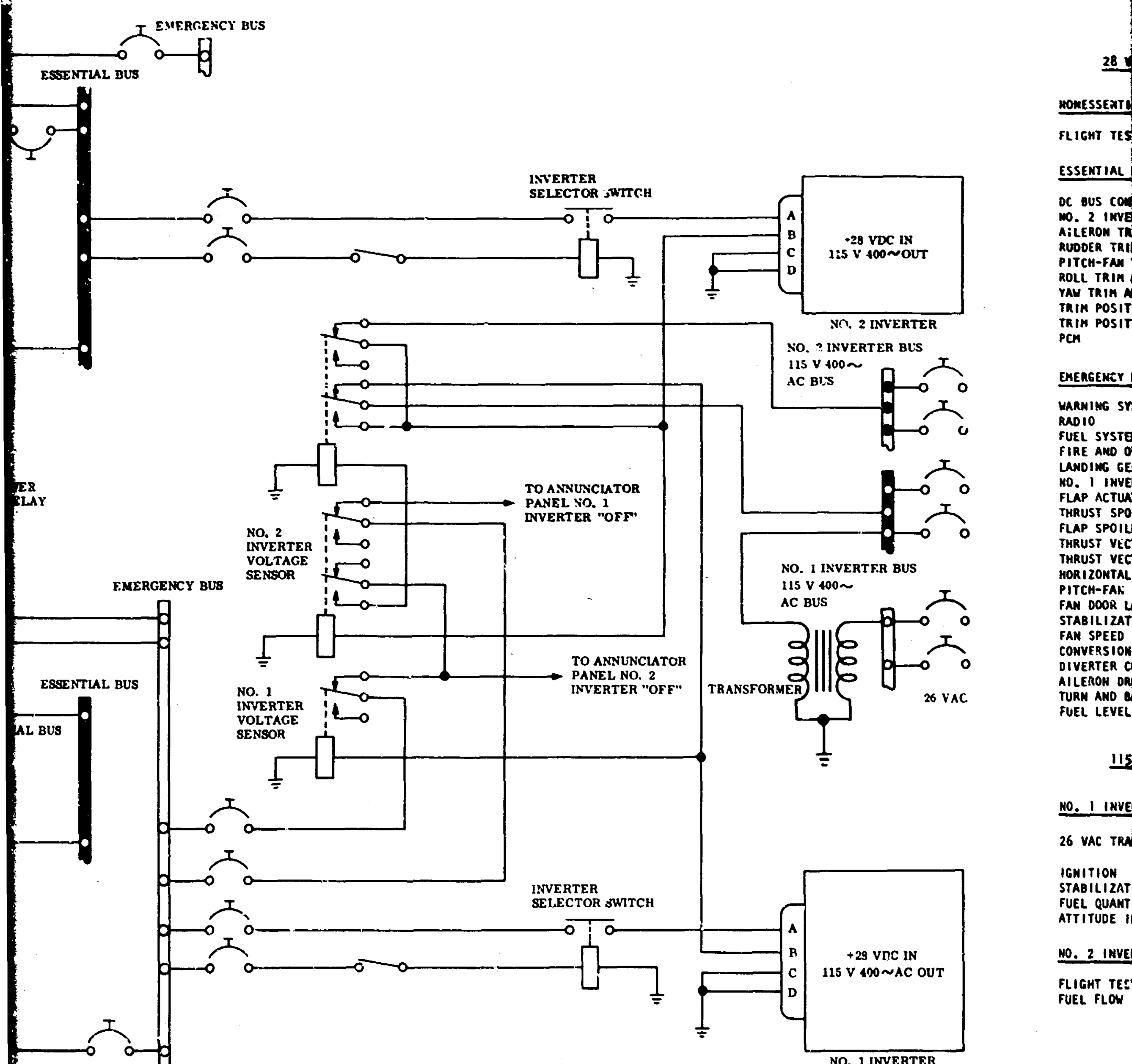
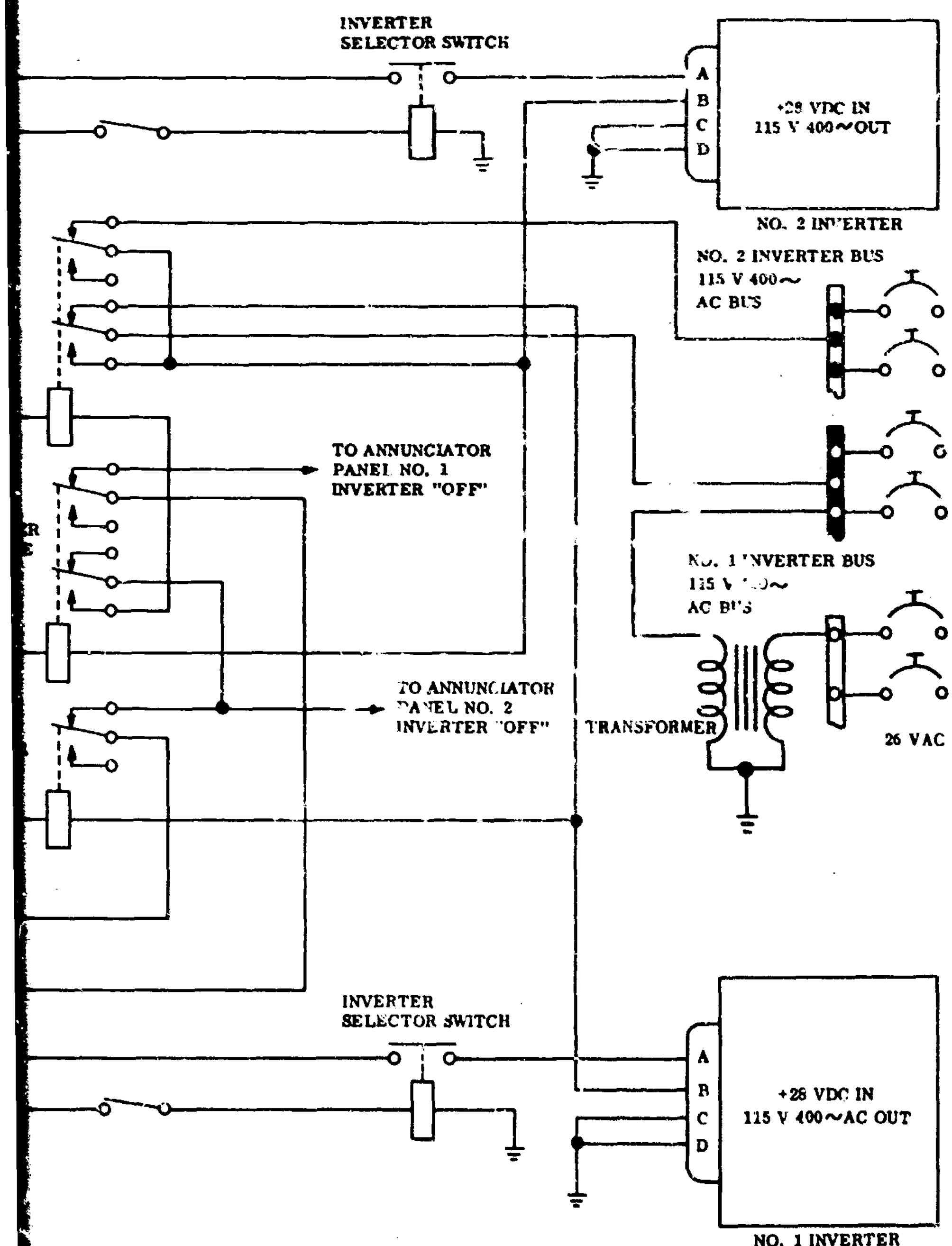


Figure 47. Conversion Control Interlock for Fan-Mode to Jet-Mode Conversion.



EMERGENCY BUS
ESSENTIAL BUS
INVERTER SELECTOR SWITCH
+28 VDC IN 115 V 400~OUT
NO. 2 INVERTER
NO. 2 INVERTER BUS 115 V 400~ AC BUS
TO ANNUNCIATOR PANEL NO. 1 INVERTER "OFF"
NO. 2 INVERTER VOLTAGE SENSOR
EMERGENCY BUS
NO. 1 INVERTER BUS 115 V 400~ AC BUS
TO ANNUNCIATOR PANEL NO. 2 INVERTER "OFF"
TRANSFORMER
26 VAC
ESSENTIAL BUS
NO. 1 INVERTER VOLTAGE SENSOR
INVERTER SELECTOR SWITCH
+28 VDC IN 115 V 400~AC OUT
NO. 1 INVERTER

B



<u>26 VDC</u>

<u>NONESSENTIAL BUS LOADS</u>

FLIGHT TEST INSTRUMENTATION

<u>ESSENTIAL BUS LOADS</u>

DC BUS CONTROL
NO. 2 INVERTER
AILERON TRIM ACTUATOR
RUDDER TRIM ACTUATOR
PITCH-FAN TRIM ACTUATOR
ROLL TRIM ACTUATOR, FAN
YAW TRIM ACTUATOR, FAN
TRIM POSITION INDICATOR, FAN
TRIM POSITION INDICATOR, CONVENTIONAL
PCM

<u>EMERGENCY BUS LOADS</u>

WARNING SYSTEM
RADIO
FUEL SYSTEMS
FIRE AND OVERHEAT DETECTOR SYSTEMS
LANDING GEAR CONTROL AND INDICATOR
NO. 1 INVERTER
FLAP ACTUATOR
THRUST SPOILER CONTROL
FLAP SPOILER POSITION INDICATOR
THRUST VECTOR ACTUATOR
THRUST VECTOR POSITION INDICATOR
HORIZONTAL STABILIZER CONTROL VALVES
PITCH-FAN INLET LOUVER ACTUATORS
FAN DOOR LATCH ACTUATORS
STABILIZATION AUGMENTATION SYSTEMS
FAN SPEED INDICATOR AND CONTROL
CONVERSION CONTROL SYSTEMS
DIVERTER CONTROL VALVES
AILERON DROOP ACTUATOR
TURN AND BANK INDICATOR
FUEL LEVEL WARNING

<u>115 VAC, 400 CPS</u>

<u>NO. 1 INVERTER BUS LOADS</u>

26 VAC TRANSFORMER { OIL PRESSURE; HYDRAULIC PRESSURE

IGNITION
STABILIZATION AUGMENTATION SYSTEMS
FUEL QUANTITY INDICATING
ATTITUDE INDICATOR

<u>NO. 2 INVERTER BUS LOADS</u>

FLIGHT TEST INSTRUMENTATION
FUEL FLOW

C

SECONDARY FLIGHT CONTROLS SYSTEM

General Description

The secondary flight controls system is made up of the following elements:

1. Jet mode:

   a. Flaps/aileron droop.

   b. Trim.

      (1) Lateral.

      (2) Directional.

      (3) Longitudinal.

   c. Wing-fan door locks.

2. Fan mode:

   a. Flaps/aileron droop.

   b. Fan overspeed cutback (J-85 power).

   c. Trim.

      (1) Lateral.

      (2) Directional.

      (3) Longitudinal.

Flaps/Aileron Droop

Single-slotted flaps are used and controlled by pilot command. Symmetrical aileron deflection is also incorporated and coordinated with flap deflection to provide 15° aileron droop when the flaps are full-down. An interlock in the flap system prevents conversion from jet-mode to fan-mode flight unless the flaps are in the full-down position. The flaps may be positioned at any intermediate point during ground operation or flight. A cockpit indicator provides flap position information.

Trim Capabilities

Lateral and directional jet-mode control force trim is accomplished by use of electrical screw jacks located in the aileron and rudder tab systems. Lateral and directional trim limits are $\pm 2^{\circ}$ aileron and $\pm 3^{\circ}$ rudder. Longitudinal jet-mode control force trim is accomplished by use of a hydraulically driven screw jack attached to the horizontal stabilizer. Longitudinal trim limits are from 5° stabilizer leading edge down to 20° leading edge up. Because of this large range of longitudinal trim capability, an emergency longitudinal trim control subsystem is provided to arrest a "runaway" stabilizer in the event of a failure in the normal subsystem.

Fan-mode control force trim is accomplished by electrical screw-jack adjustment of stick and rudder pedal positions for zero control centering spring forces. Preprogrammed trim of the nose-fan thrust modulators during transition, as a function of exit louver position (vector angle), is provided by the mechanical mixer. Automatic trim of the horizontal stabilizer for conversion is programmed by the conversion control interlock system through electrical control of the stabilizer actuator position and actuation rate.

Wing-Fan Door Locks

Wing-fan inlet door locks are provided to ensure safe jet-mode flight in the event that both hydraulic systems fail.

Automatic Throttle Cutback System

Fan overspeed protection is provided by an automatic throttle cutback system. The primary purpose of this system is to prevent destructive fan overspeeding that could result from fan stall caused by high power, high speed (in fan mode), or high angle-of-attack conditions, the onset rate of which would be beyond the pilot's ability to respond with manual power reduction. The system monitors wing- and nose-fan rpm. The system is armed at 100-percent wing-fan rpm and 104-percent nose-fan rpm; it lights the annunicator fan overspeed and master caution lights. When either 103-percent wing-fan rpm or 110-percent nose-fan rpm is exceeded, a pneumatic actuator drives the throttle linkages (but not the throttle handles or lift stick twist grip) down to 70-percent J-85 power settings ($\cong$97-percent rpm) on both engines. The pilot may override the cutback with a power reset switch from either the twist grip position or the throttle quadrant position. However, the reset will not hold until wing-fan rpm has been reduced below $\cong$98 percent. Normally, additional manual J-85 engine power reduction is needed; approximately 40 pounds of pilot effort is required to move the throttles to the cutback point at which the brake

effect is released and at which the necessary reduction can be accomplished. After fan rpm has decreased sufficiently, a power reset command will then hold in.

MAINTAINABILITY HISTORY AND RECOMMENDATIONS

Failure Rate

Seventeen control system failures were reported. Four of these, which are described below, were reported during a flight or ground test mission. The remaining 13 failures were reported during the various inspection and functional test procedures. The man-hours expended in correcting discrepancies are shown in Table IX.

Thrust Vector Actuator

This component in the flight controls system had the highest failure rate. Special maintenance procedures for the actuator were established to ensure adequate component performance. Based on reported vector actuator failures, the most likely types of component failure would only result in delays, aborts, or terminations. However, fan-mode transition envelope expansion or high-performance transition missions may require added precautions to minimize thrust vector actuator failure risks. The precautions should include (1) verification of proper actuator (vectoring) rate and actuation continuity (that is, no intermittencies) and (2) pilot preflight briefing to remind him of the effects of the most likely vector actuator modes of failure on transition function programming and transition maneuvering.

Thrust vector actuator operation without hydraulic power on the aircraft is prohibited. However, accidental ground operation has occurred; as a result, both the actuator and the control system linkage have been damaged. It is recommended that a study be conducted to evaluate (1) positive methods of preventing actuator operation when electrical power, but not hydraulic power, is available on the aircraft and (2) the effects of these methods on related systems and components. It is further recommended that this study include evaluation of parallel hydraulic pressure switches (that is, one in each system) in the vector actuator power circuit as a means of corrective action.

System and component detail requirements for derivative aircraft will depend on operational requirements specified for a given configuration. It is recommended that future requirements for XV-5A-type collective vector controls be identified specifically as a primary flight control

TABLE IX. MAN-HOURS EXPENDED IN CORRECTING DISCREPANCIES IN FLIGHT CONTROL SYSTEM

Part Name	Preparation	Diagnosis	Accessibility	Corrective Action	Reassembly	Checkout	Total Man-Hours	Remarks
Throttle Cutback Valve	0.1	2.0	0.5	0.9	0.1	0.5/0.2*	4.1	Waugh Box checkout required
Link of Mechanical Mixer	0.1	0.1	0.2	0.3	0.2	0.1	1.0	Straightened
Thrust Vector Actuator	0.1	0.2	0.5	3.0	0.5	2.0/1.0	6.3	Replaced
Stability Control Actuator	2.0/1.0	0.3	1.0	8.7/3.7	1.0/0.5	1.0/0.5	14.0	Bushings and bearing replaced
Pitch Control Door Damper	0.2	0.1	0.2	0.5	0.5	0.2	1.7	Replaced
Cover Door Latch	0.1	0	0.2	0.5	0.2	0	1.0	Crack welded
Rod End Actuator Fan Inlet Door	0.2	0.1	0.2	0.2	0.3	0.1	1.1	Replaced - excessive looseness
Bearing Inboard Flap Fitting	0.3	1.2	0.1	2.0	0.3	0.2	4.1	Replaced - excessive looseness
Actuator, Mechanical Mixer Assembly	0.5	0.5	1.0	4.0	1.0	1.0	8.0	Replaced - position indicator pot was open
L.H. and R.H. Thrust Spoiler Door	12.0/8.0	0	2.0	2.0	3.0/1.5	0.3	19.3	Numerous cracks welded
Clip Thrust Spoiler and Fairings	0.2	0	0.2	1.0	0.5	0.1	2.0	Replaced
Actuator New Mechanical Mixer Assembly	0.2	0.5	1.0	3.0	1.0	0.3	6.0	Rewired

*If 2 numbers are divided by a slash, the first number represents the man-hours; the second, the clock hours.

function (as opposed to a trim function). The following items should be evaluated when vector actuator requirements are defi.

1. Dual-Drive Capability

   The dual-drive capability includes (1) simultaneous independent redundant drive, (2) primary/standby drive with automatic changeover, and (3) primary/standby drive with manual changeover.

2. Dual Wiring

   If an electric actuator is used, dual wiring to the last possible termination points inside the actuator should be considered.

3. Duty/Life Cycle Requirements

   Criteria to establish minimum acceptable duty/life cycle requirements should be defined.

Pilot-Induced Lateral Oscillations

During low-speed conventional flight, a tendency toward pilot-induced lateral oscillations has been observed. New pilots appear to learn to overcome this tendency quickly. This condition is attributed to the combination of high lateral control power and reduced effectiveness of the aileron tabs at low speeds, which results in both low stick centering forces and a low force gradient.

It has been suggested that removal of the fan-mode feel and stick centering spring package, incorporation of feel and stick centering springs in the aileron controls, and reduction of the aileron tab gearing ratio would not only improve low-speed lateral control but would also tend to linearize (and thus to improve) the system on up through the high-speed range. Such a system would also retain the present desired fan-mode characteristics. It is recommended that this suggestion be evaluated for incorporation in the XV-5A and that it be considered for application to derivative aircraft.

Engine Power Control

The lift stick twist grip requires approximately a 345° rotation to move the engine control from the stop position to the 100-percent rpm position. This has resulted in a somewhat more awkward power control than is considered to be optimum, particularly when large power changes are desired, such as during a hovering landing. Handling qualities may be improved by a modification in the effective gear ratio of this control by

retaining present sensitivity for normal power changes and by increasing sensitivity for large power changes.

Fan Overspeed Power Cutback System

The fan overspeed power cutback system appears to satisfy the basic requirement, which is to provide protection against destructive fan overspeed. However, a number of undesirable conditions associated with this system have developed, among which are the following:

1. Desensitized pilot response to annunciator panel and master caution signals due to frequent occurrence of a fan overspeed condition during normal transition flight.

2. Performance degradation caused by the need either to reduce engine power sufficiently to reduce fan rpm below the overspeed warning point to clear the cutback and the warning light, or to maintain sufficiently low power settings to prevent warning or cutback occurrence.

3. Pilot dissatisfaction with manipulation requirements.

4. Change in maximum continuous speed allowable for rpm limit from 100 percent to 103 percent without appropriate change in warning system, which results in a false master caution signal during normal operation.

5. System sensitivity to warning or cutback initiation during normal operation as a result of minor changes in aircraft speed/attitude, vector angle, or power setting.

6. Single actuator powered by emergency pneumatic system.

The requirement for positive, automatic fan overspeed protection under adverse flight conditions must be met. A thorough study of this system is recommended; it must include full consideration of flight safety, associated components reliability, human factors, associated aircraft handling qualities, and aircraft performance.

DESIRABLE FEATURES

1. In the jet mode with both hydraulic systems inoperative, the aircraft is designed to be controllable without the use of automatic or manual changeover features.

2. The conversion interlock system is designed such that no single failure will result in an uncontrollable aircraft during conversion.

3. The pilot can always reverse the sequence at any point during the conversion to bring the aircraft back to the flight configuration that it was in prior to the start of conversion.

UNDESIRABLE FEATURES

1. Low lateral stick centering force and force gradient during low-speed jet-mode flight.

2. Reduced lateral control authority with increasing lift control commands (in mechanical mixer).

3. Twist grip (on lift stick) throttle sensitivity.

4. Thrust vector actuator.

5. Fan overspeed power cutback system.

ELECTRICAL SYSTEM

SYSTEM CONFIGURATION AND OPERATION

Power Generating System

The 28-volt dc primary aircraft power is supplied by two brushless generators with internal regulators. Each generator is driven by one of the J-85 gas turbine engines. Provisions are made for load sharing between the two generators. A control panel, located in the electronics compartment, is provided for each generator for protection against overvoltage, for feeder fault protection, for bus protection by reverse current rectifiers, for automatic load monitoring, for automatic equalizer disconnect, for system deenergizing, and for trip-free resetting (see Figure 48). One bus control panel is also installed in the electronics compartment.

The total system capability is 9 kilowatts, with a load requirement of 2.8 kilowatts. These loads are divided into three categories: (1) emergency loads, (2) essential loads, and (3) nonessential loads.

Generator fault detectors described above are provided so that if either generator fails, the nonessential bus is automatically removed from the circuit. Should both generators fail, both the essential and the nonessential buses would drop out, leaving the battery-operated emergency bus to operate the remaining aircraft circuits. Cockpit indicators warn the pilot of generator failures and automatic bus load clearing. An external 28-volt dc ground power connection is provided for aircraft starting, ground servicing, and checkout purposes.

Power Control System

The 115-volt ac, 400-cps power is provided by two 250-volt-ampere rotary inverters. The No. 1 and No. 2 inverters are driven by the 28-volt dc emergency bus and essential bus, respectively. Since the emergency bus is supplied by the battery if both generators fail, items on the No. 1 inverter will remain in operation while items on the No. 2 inverter will be lost. The 115-volt ac essential loads and the 115-volt ac nonessential loads are fed from buses marked accordingly. If either inverter fails, the 115-volt ac nonessential bus drops out and the 115-volt ac essential bus is powered by the remaining inverter. The 26-volt ac, 400-cps bus power is provided by a transformer on the 115-volt ac essential bus.

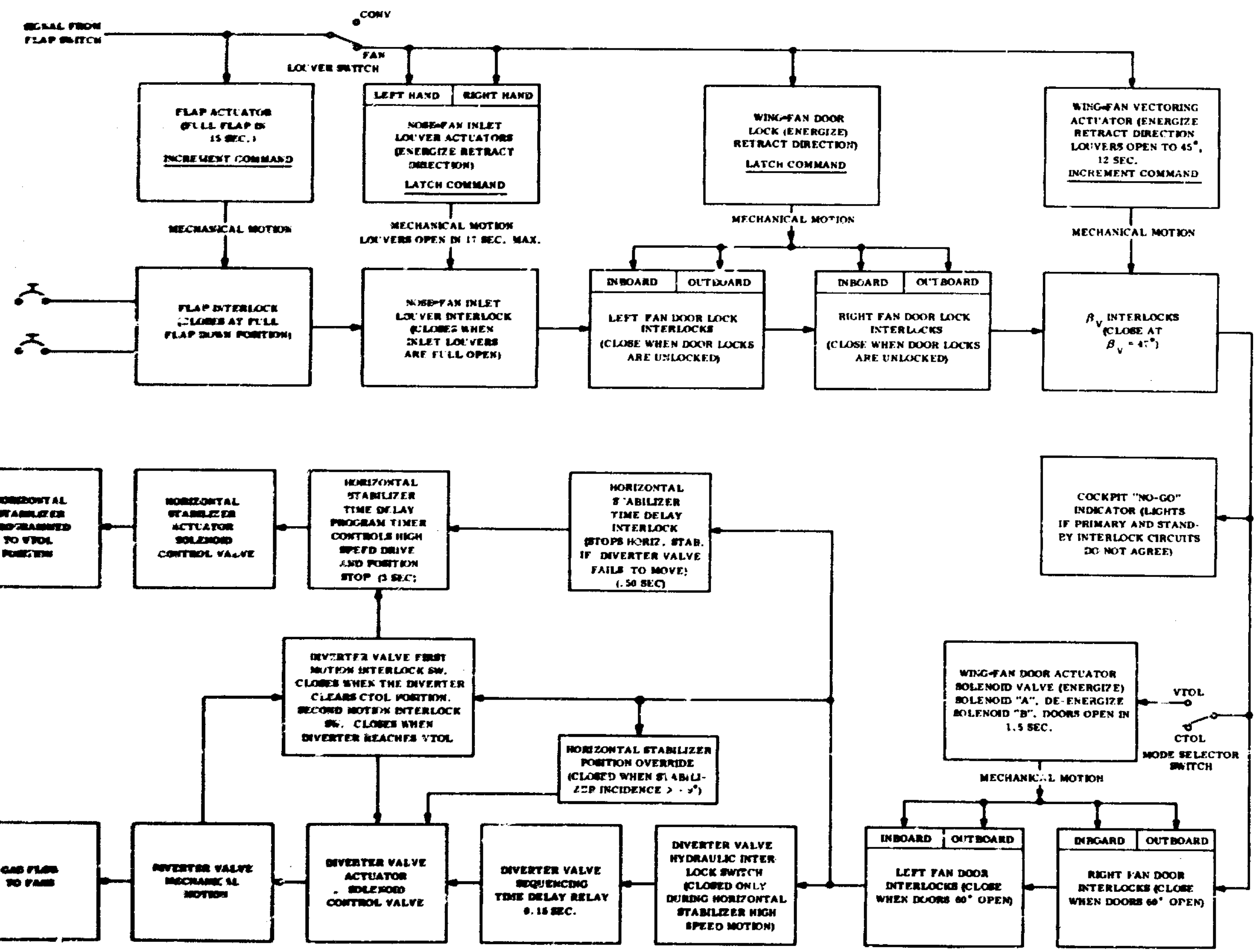


Figure 48. Electrical Power System.

Cockpit indicators warn the pilot of inverter failures. External 115-volt ac, 400-cps power is utilized for ground servicing and checkout.

The emergency battery is a 28-volt dc (nominal), 25-ampere-hour, silver-zinc type, which, after being derated for 9 months' maximum service life, can provide the required 5 minutes of emergency bus current at 60 amperes plus one conversion (mode change) and still maintain sufficient capacitance for good ripple regulation. When the aircraft is in the conventional configuration, maneuvering flight can be sustained without any electrical power.

Power Distribution System

Power is distributed to all systems, subsystems, and components by open wire harness construction. Where possible, harnesses are separated with respect to their functions. A schematic of the power distribution system is shown in Figure 49. One circuit-breaker panel is installed on the cockpit floor on the right side of the pilot's seat. Three ac buses (two 115-volt, 400-cps and one 26-volt, 400-cps) and two dc buses were installed as part of the circuit-breaker panel. The circuit-breaker panel consists of approximately 100 circuit breakers, the largest value of which is 25 amperes. All circuit breakers are of a size compatible with the current-carrying capacity of their distribution wires.

The CCIS comprises the bulk of the aircraft electrical distribution network.

MAINTAINABILITY HISTORY AND RECOMMENDATIONS

Failure Rate

Eighteen electrical system installation failures were reported; of these, seven were reported during a flight or ground test mission. Table X shows the man-hours expended in correcting discrepancies.

Generator and Control Panel

These two components indicated the same highest electrical system failure rate. Poor accessibility to the generator regulator for adjustment and difficult generator removal and installation procedures required excessive maintenance man-hours. In addition, use of Glyptal to secure both regulator and control panel adjusting potentiometers is unsatisfactory and contributes to this problem. Relocating or reorientating the adjusting potentiometer and providing for positive locking have been suggested. Improvements in generator mounting to facilitate rapid removal and replacement have also been suggested. It is recommended that detail specifications

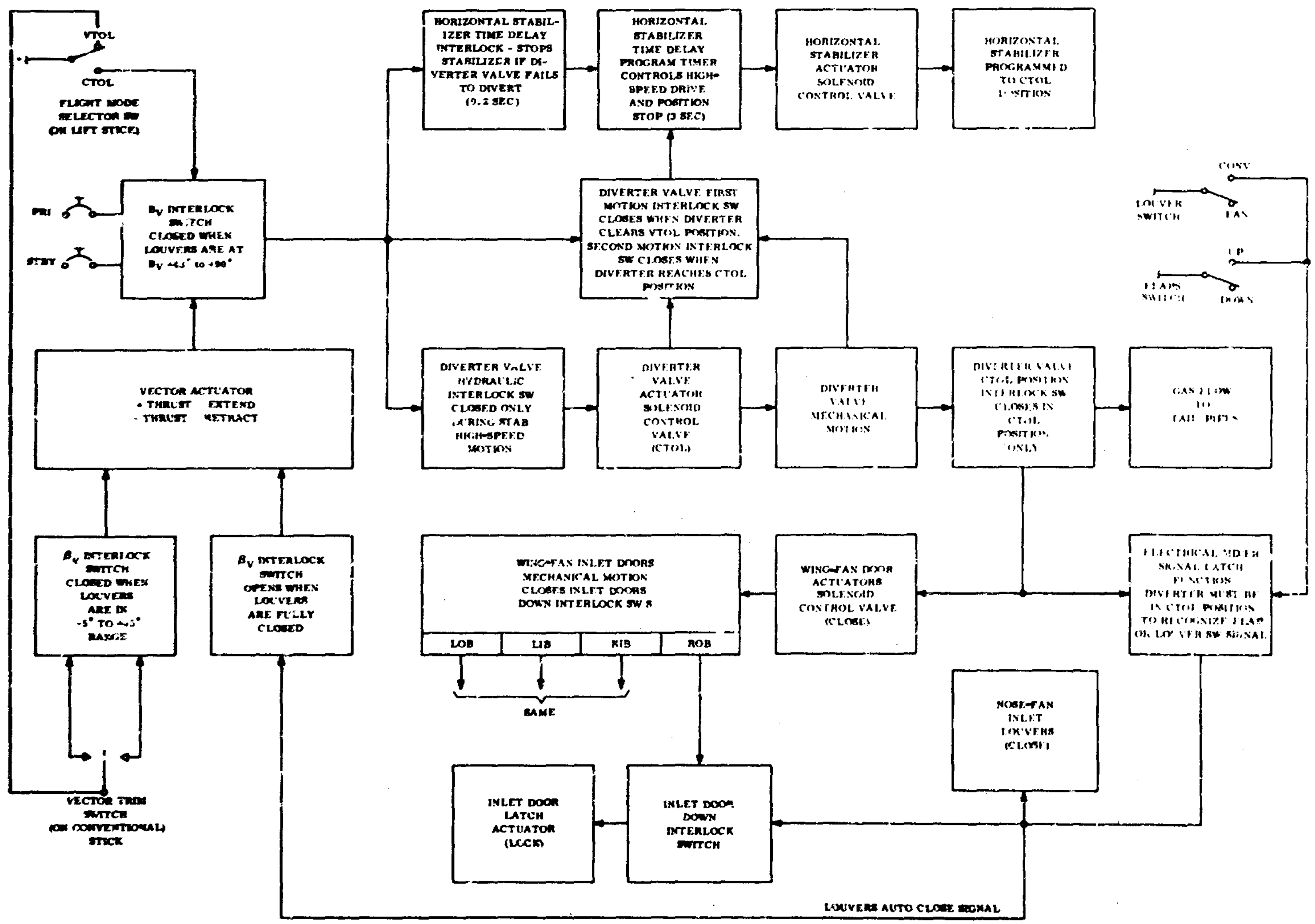


Figure 49. Electrical Power Distribution System.

TABLE X. MAN-HOURS EXPENDED IN CORRECTING DISCREPANCIES IN ELECTRICAL SYSTEM

Part Name	Preparation	Diagnosis	Accessibility	Corrective Action	Reassembly	Check-out	Total Man-Hours	Remarks
Waugh Box	0.2	0.1	0.1	4.8	0.1	0.4	5.7	Replaced
EGT Harness	0.4/0.2*	0.2/0.1	1.0/0.5	6.7/3.7	1.0/0.5	2.0/1.0	11.3	Plug replaced
Linear Accelerometer	0.1	0.1	0.3	0.5	0.4	0	1.4	Replaced
Generator Control Panel	0.5	2.0	0.5	3.0	1.0	2.0/0.5	9.0	Replaced; radio buzz traced to gen. system
Generator	2.0	40.0/8.0	2.0/1.0	36.0/24.0	2.0/1.0	6.0/2.0	88.0	Replaced; radio buzz traced to gen. system
Inverter	0.3	0.2	0.2	1.0	0.3	0.1	2.1	Replaced; improper fuel-flow reading
Wiring Harness (runs through space frame)	1.0	1.0	1.0	22.0	2.0	1.0	28.0	Rerouted and spliced
Wing-Fan Door Lock Actuator	0.1	0.2	0	1.0	0	1.0/0.5	2.3	Special rig check tool required
Wing-Fan Door Actuator Safety Key	0.5	0.1	0.1	0.1	0.5	0.2	1.5	Key replaced
Wing-Fan Door Lock Actuator	0.1	0.2	0	1.5	0	0.5	2.3	Replaced
Wing-Fan Door Lock Actuator	0.2	0.3	0.2	6.0	1.1	0.2	8.0	Replaced
Wing-Fan Actuator Fairing	0.1	0.1	0.2	0.8	0.2	0	1.4	Plastic repaired
Wind-Fan Door Lock Actuator Cover	0.1	0.1	0.2	0.5	0.2	0.1	1.2	Crack welded
Wing-Fan Inlet Door Actuator Arm	0.1	0.2	0.1	0.6	0.1	0.1	1.2	Replaced

*If 2 numbers are divided by a slash, the first number represents the man-hours; the second, the clock hours.

for derivative aircraft include suitable criteria and requirements to preclude or to minimize these types of problems.

Inverters

Three inverter failures have been attributed primarily to high temperatures in the aft equipment compartment. Improved environmental control for this compartment is discussed in the cooling system section of this report.

DESIRABLE FEATURES

1. The electrical power system is designed such that the failure of one or both generators will not result in loss of emergency bus power (28-volt dc).

2. Jet-mode maneuvering flight can be sustained without any electrical power available.

UNDESIRABLE FEATURES

1. Generator and control panel require excessive maintenance.

2. Inverter hot environment causes excessive inverter failures.

3. Battery maintenance is excessive.

4. Location of circuit-breaker panel permits personnel to step and walk on it.

HYDRAULIC SYSTEM

SYSTEM CONFIGURATION AND OPERATION

General Description

The hydraulic system consists of two completely independent systems (see Figure 50). Each system pump is engine-driven, system No. 1 being driven by the left-hand engine and system No. 2 by the right-hand engine. Both systems are completely enclosed; bootstrap reservoirs are utilized for pressurization of the pump suction lines.

The two systems power 17 conventional cylinder-type actuators, 7 servo actuators, and 1 hydraulic motor-driven screw jack. These are discussed below.

All primary and secondary flight control hydraulic actuators are tandem except the landing gear actuators and the thrust spoiler actuator. During normal operation, the tandem actuators derive half of their operating power from each hydraulic system.

A dual pressure gage is located on the instrument panel to monitor hydraulic system pressures at all times. In addition, low-pressure warning lights on the annunciator panel are provided to indicate pressure loss in either system. External ground power connections are provided to facilitate servicing and systems checkout.

Hydraulic system component locations are shown in Figure 51.

Thrust Spoiler Actuator

The thrust spoiler actuator is powered by hydraulic system No. 1 only; internal locks are provided to lock the spoilers in the retracted position in the event of system pressure loss.

Landing Gear Control Actuator

The landing gear control actuators are also powered by hydraulic system No. 1 only; however, the emergency pneumatic system provides for emergency extension.

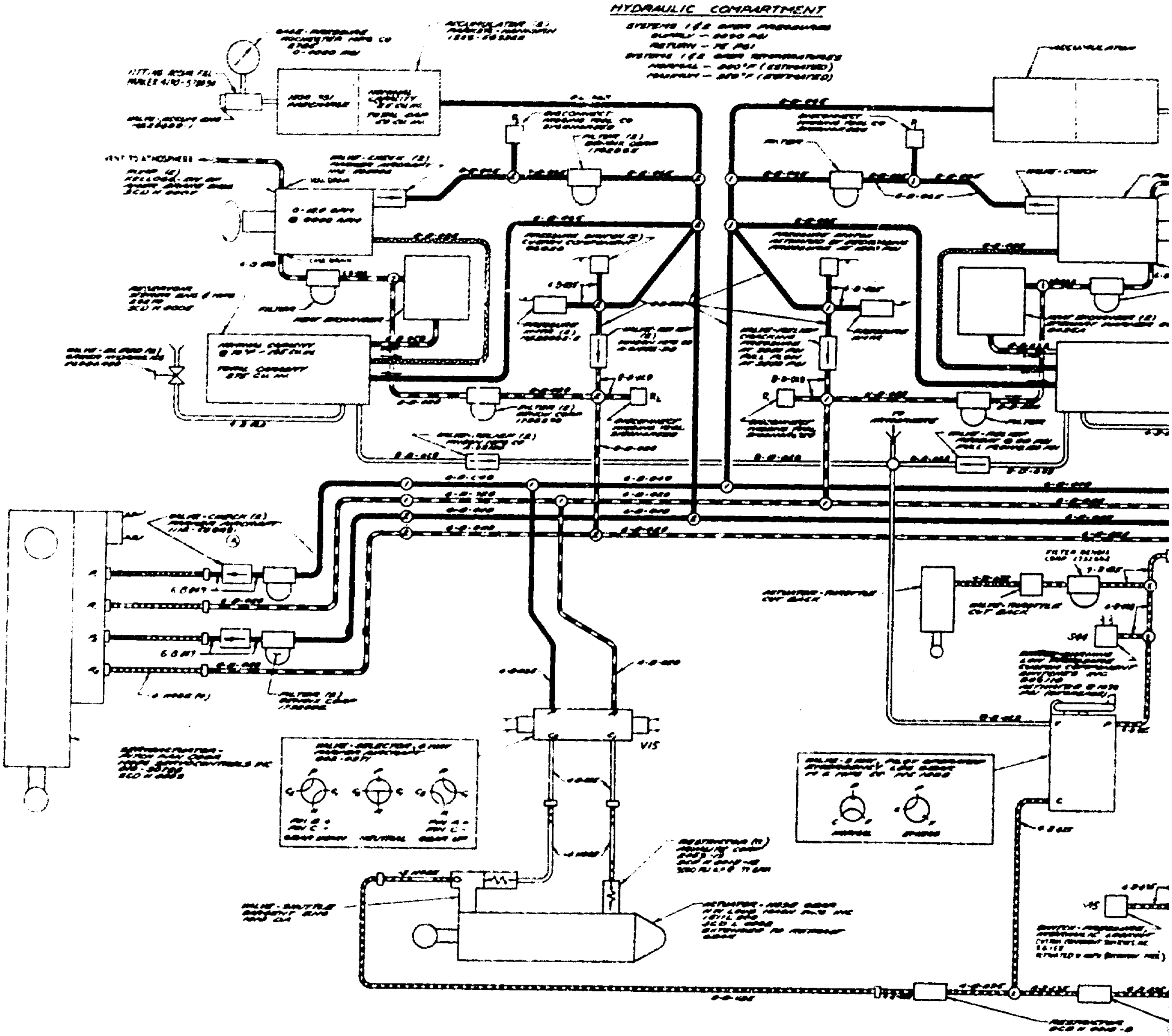


Figure 50. Hydraulic and Pneumatic Schematic.

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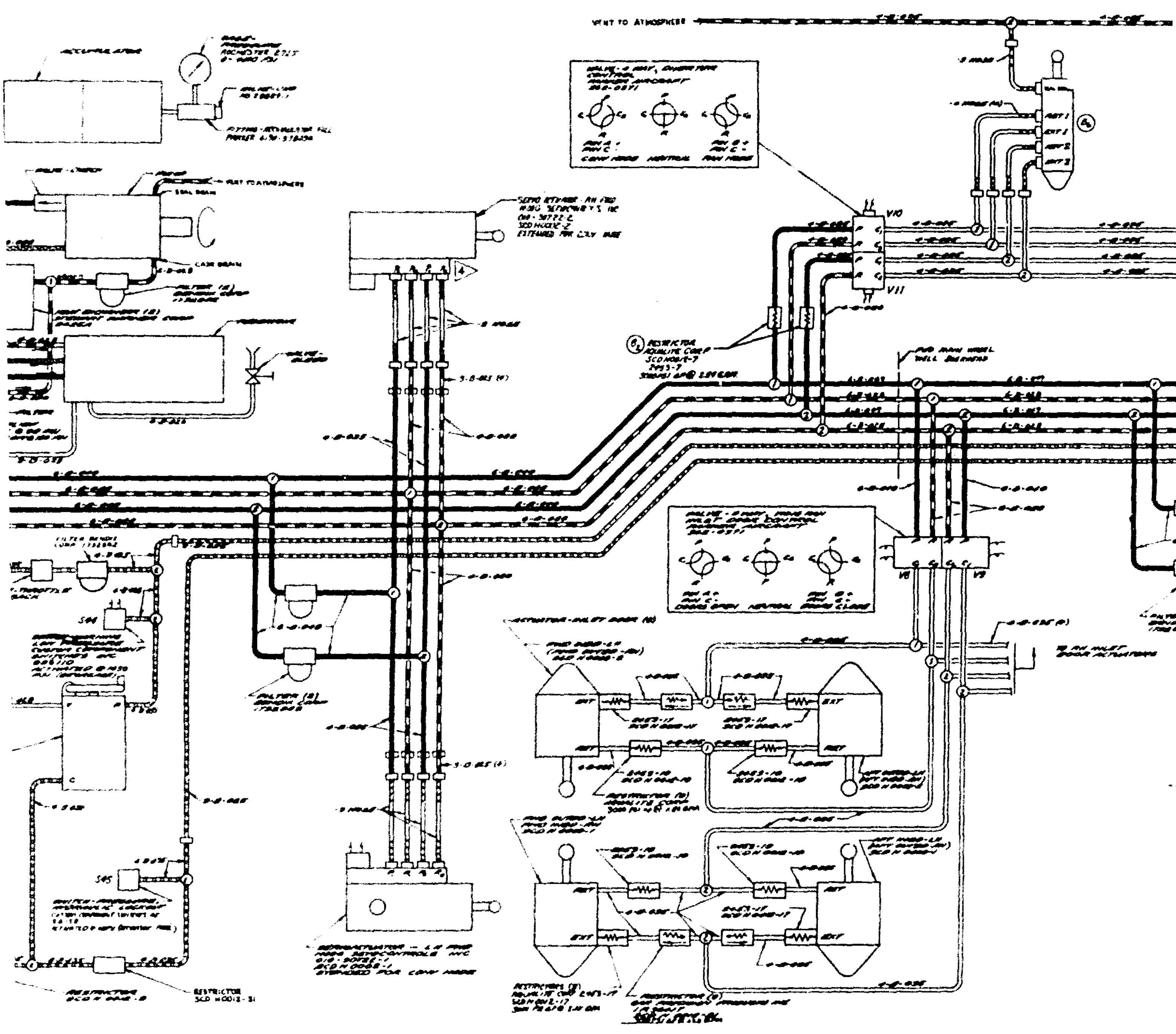
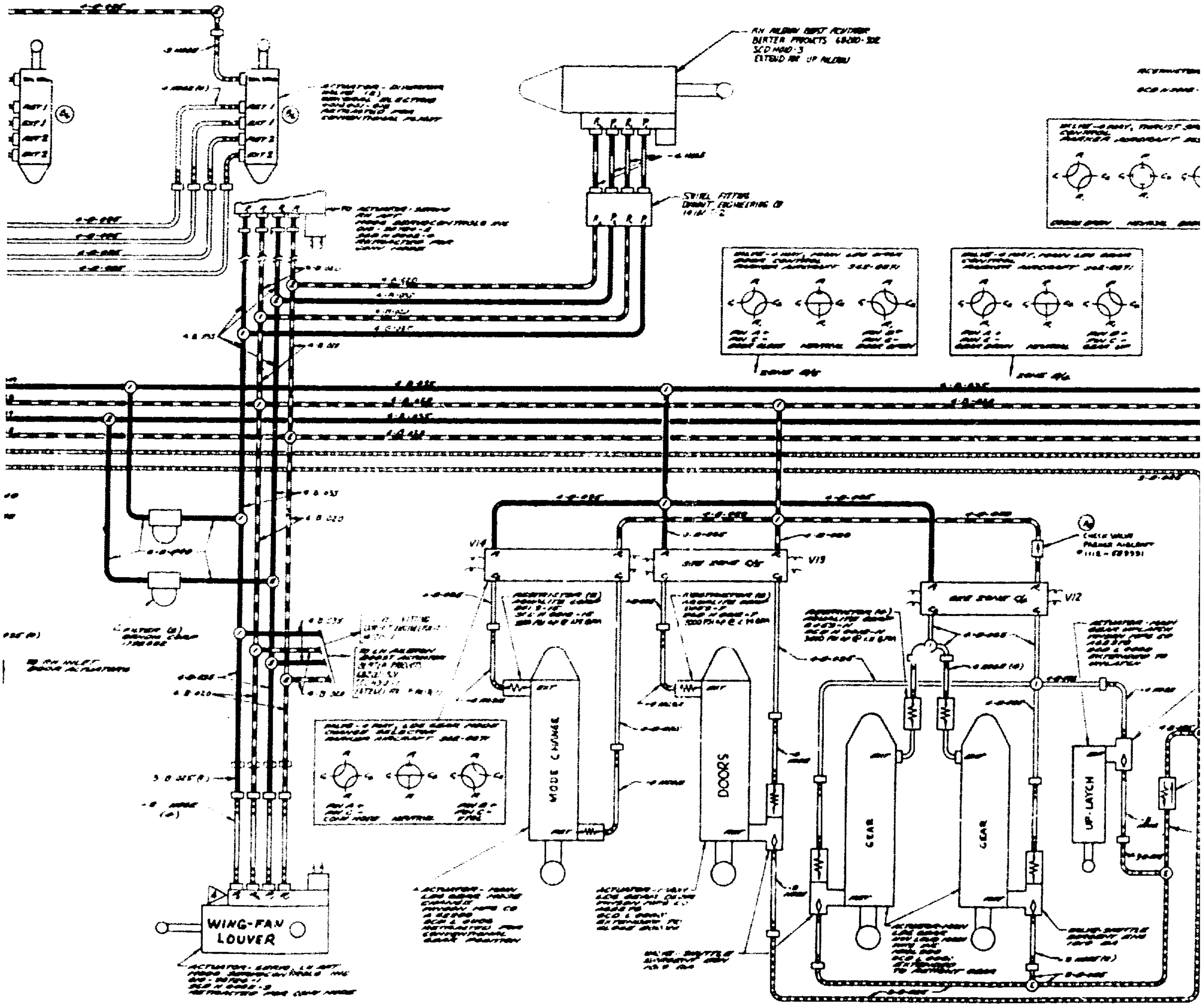
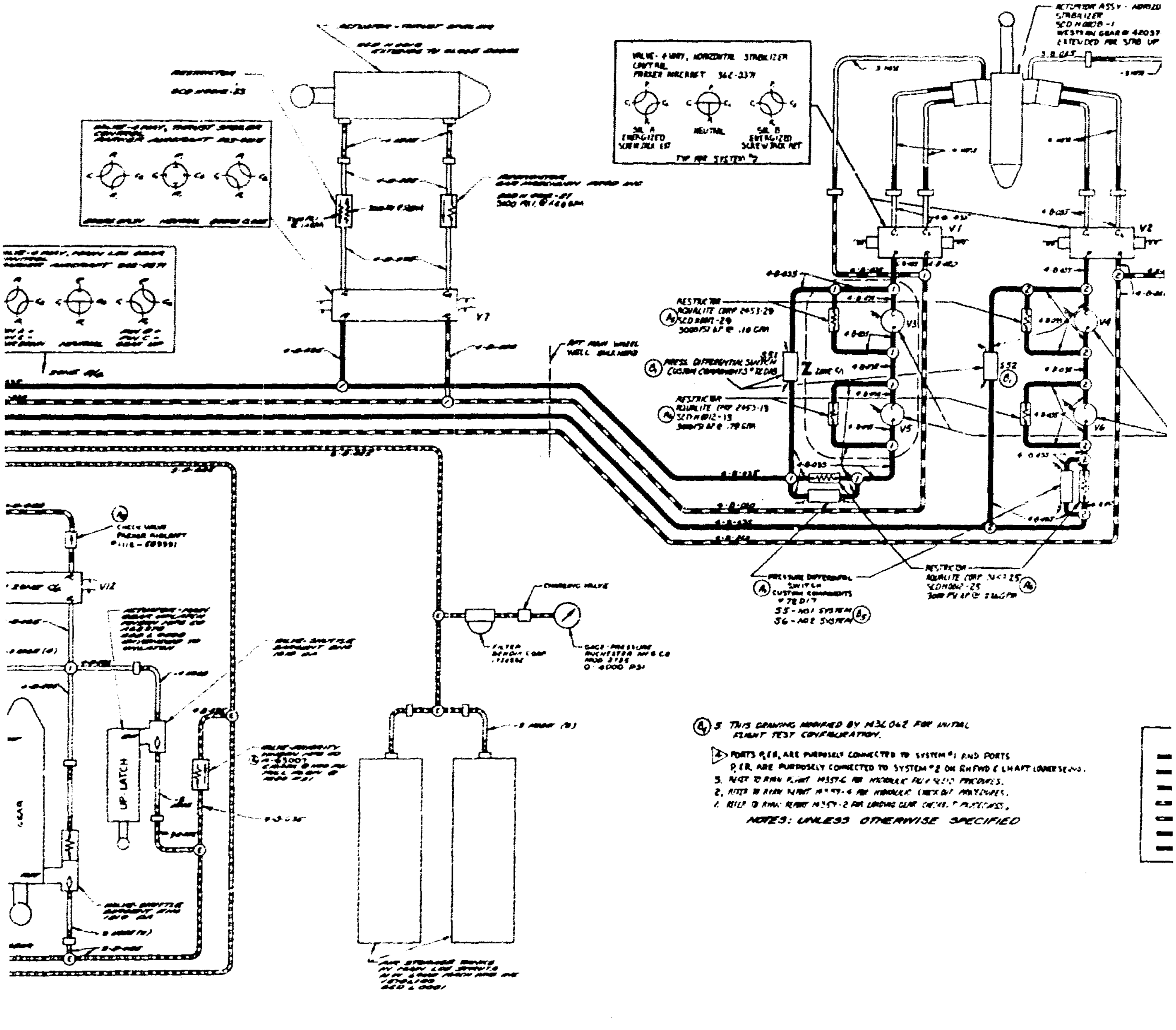
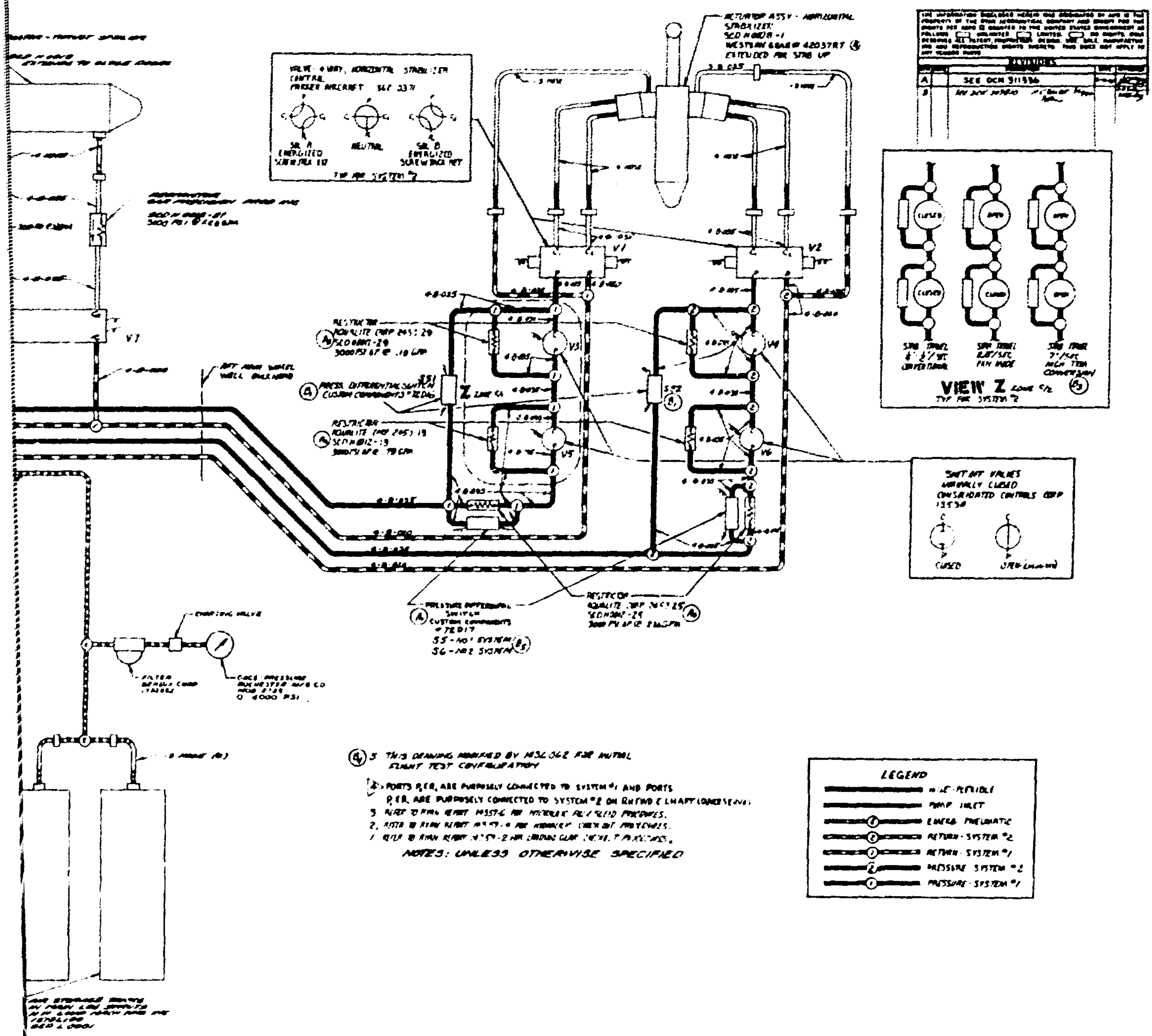

VENT TO ATMOSPHERE
V10
V11
V8
V9
S44
S45
RESTRICTOR
SCD H0012-31



SWIVEL FITTING
V14
V13
V12
MODE CHANGE
DOORS
GEAR
GEAR
UP-LATCH
WING-FAN
LOUVER



VALVE - 4 WAY, HORIZONTAL STABILIZER CONTROL
TYP FOR SYSTEM #2
V1
V2
V3
V4
V5
V6
V7
V12
UP-LATCH
RESTRICTOR
PRESS. DIFFERENTIAL SWITCH
PRESSURE DIFFERENTIAL SWITCH
S5 - NO 1 SYSTEM
S6 - NO 2 SYSTEM
CHARGING VALVE
FILTER
GAGE - PRESSURE
0 - 4000 PSI
CHECK VALVE
THIS DRAWING MODIFIED BY M3L062 FOR INITIAL FLIGHT TEST CONFIGURATION.
PORTS P, CR, ARE PURPOSELY CONNECTED TO SYSTEM #1 AND PORTS P, CR, ARE PURPOSELY CONNECTED TO SYSTEM #2
NOTES: UNLESS OTHERWISE SPECIFIED



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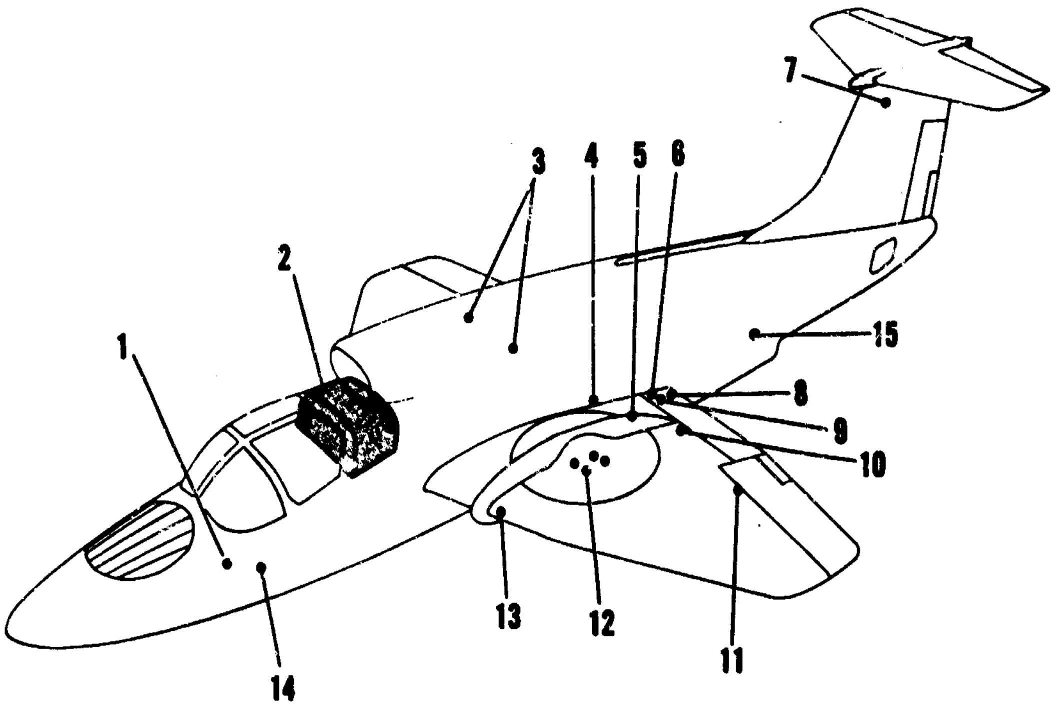


1. Thrust Modulator Door Actuator
2. Hydraulic Equipment Compartment
3. Diverter Valve Actuator
4. Right-Hand Main Landing Gear Retract Actuator (MLG Wheel Well)
5. Left-Hand Main Landing Gear Retract Actuator (MLG Wheel Well)
6. Main Landing Gear Uplock Actuator
7. Horizontal Stabilizer Trim Actuator
8. Main Landing Gear Door Actuator (left-hand side only) (MLG Wheel Well)
9. Main Landing Gear 2-Position Actuator (on centerline of airplane) (MLG Wheel Well)
10. Left-Hand Aft Exit Louver Actuator
11. Left-Hand Aileron Boost Servo Actuator
12. Left-Hand Main Lift Fan Inlet Door Actuator (4)
13. Left-Hand Forward Exit Louver Actuator
14. Nose Landing Gear Retract Actuator (Nose Wheel Well)
15. Thrust Spoiler Actuator

Figure 51. Hydraulic System Component Location Diagram.

Conventional Cylinder-Type Actuators

The 17 conventional cylinder-type actuators control nose and main gear extensions and retraction, main gear door position, main gear uplatch, main gear aircraft operating mode position, diverter valve position, wing-fan inlet door position, and thrust spoiler position.

All conventional cylinder-type actuators are controlled by three-position, four-way, solenoid-operated control valves (see Figure 52). These valves are pressure-operated through solenoid-operated poppets. If solenoid No. 1 is energized, return pressure will be applied to the left-hand end of the valve spool. Since the right-hand end is subjected to full system pressure, the valve spool will displace to the left, thereby retracting the cylinder. Similarly, if solenoid No. 2 is energized when solenoid No. 1 is deenergized, the cylinder will extend. The valve spool is spring-loaded at each end; sufficient preload assures that the spool will remain centered if both solenoids become energized because of an electrical system failure. Since both cylinder chambers which are controlled by that valve are interconnected when the valve is centered, a "hydraulic lock" is prevented and the system No. 2 valve will still control the cylinder properly.

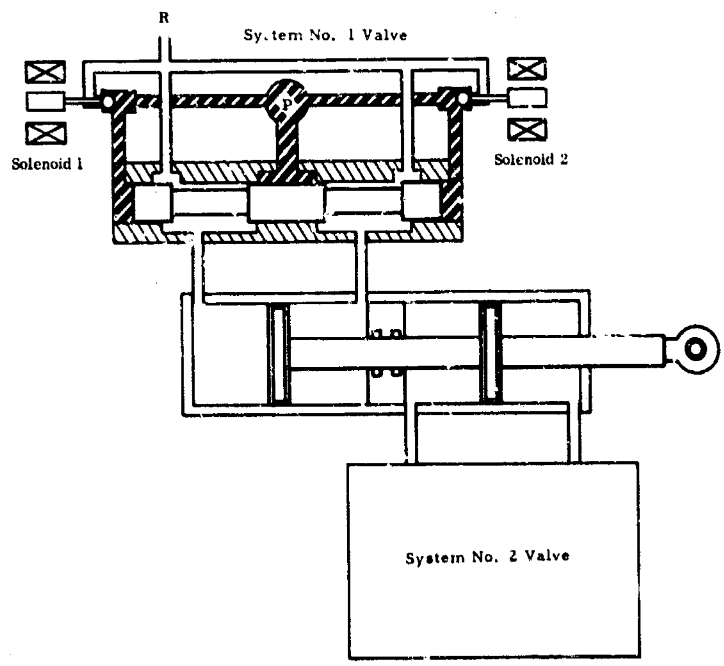


Figure 52. Hydraulic System, Three-Position, Four-Way Control Valve Schematic.

Operating rates of the conventional cylinder-type actuators are controlled by restrictors in the control lines (see page 126).

Servo Actuators

The seven servo actuators control the wing-fan exit louver position and the nose-fan thrust modulator door position, and they provide the aileron boost function. The five tandem hydraulic actuators driving the exit louvers and thrust modulator doors are controlled by dual-input-type servo valves which superimpose electrical commands from the SAS upon the mechanical pilot commands. Response to electrical inputs is limited to approximately 25 percent of the cockpit control authority; sufficient control authority is reserved for the pilot to overcome SAS hard-over command failures and to maintain limited control of the aircraft.

Operating rates of the seven servos are controlled through the servo spool positions, which, in turn, are controlled by the cockpit control command rates. Thus, the servo response rates will be proportional to the control command rates up to the saturation velocity of the servos. The saturation velocities are approximately 100° per second for the wing-fan exit louvers and ailerons and 400° per second for the nose-fan thrust modulator doors.

Time Rates for Hydraulically Controlled Actuators

The normal rates for the hydraulically controlled actuators are as follows:

Nose gear	To extend, 4 seconds; to retract, 4 seconds.
Main gear	To open door, 3 seconds (doors must be full-open before priority valve permits gear-extend actuators and gear-uplock-release actuator to become pressurized).
	To extend, 5 seconds; to retract, 5 seconds.
	To change mode, 7 seconds; time to door closure after gear full retract (uplock), 3 seconds.
Wing-fan inlet doors	To open, 1.5 seconds; to close, 3 seconds.
Diverter valves	Time to fan-mode position, 0.4 second; time to jet-mode position, 0.4 second.
Thrust spoiler	To extend, 8 seconds; to retract, 2 seconds.

Motor-Driven Screw Jack

The motor-driven screw jack provides longitudinal trim by controlling the horizontal stabilizer position. It is driven at three different speeds, depending upon the flight condition at the time of operation. In the VTOL mode of operation, the tail is driven at approximately 3° per second. In the conventional mode, it is driven at approximately $1/3^{\circ}$ per second; during a conversion, it is driven at approximately $7\text{-}1/2^{\circ}$ per second to a preset tail incidence angle.

Speed is automatically controlled through a valve-restrictor combination that utilizes three restrictors in series to achieve the low rate, two in series to achieve the intermediate rate, and only one for the high rate. The number of restrictors is reduced from three to two or one by energizing two-position valves to bypass the appropriate restrictor(s).

Pitch Axis Stability Augmentation

Hydraulic system No. 1 supplies power to the secondary controls and to the first stage of the servo valve for the nose-fan thrust modulator doors. If system No. 1 becomes inoperative, no pitch axis stability augmentation will be available electrically. To reduce the probability of loss of pitch axis stability augmentation, the first stage of this servo valve should be powered by hydraulic system No. 2, which operates primary flight control only.

Restrictors

The restrictors used to provide hydraulic actuator rate control are small, lightweight units. Two nonremovable filter screens on each side of the orifice prevent blockage by system contaminants (primarily rubber particles from 0-rings). The filter capacity is relatively small, and cleaning is impractical because the particles become trapped between the screens. Satisfactory service life of restrictors in the XV-5A has been achieved primarily because of the low system contamination level due to required maintenance procedures and to the nature of the ground and flight test program.

Horizontal Stabilizer

The horizontal stabilizer is actuated by two hydraulic motors driving an integral self-locking screw jack. Each motor is operated from one of the primary hydraulic systems through control valves, bypass valves, and flow restrictors. In conversion from fan-powered to conventional flight, or vice versa, the stabilizer is automatically programmed at its maximum rate to a predetermined optimum angle for the particular mode of flight.

Limit switches at this point actuate the motor control valves to the closed position, thus stopping the stabilizer and deactivating the automatic transition programming. Thereafter, the pilot may trim the stabilizer in the conventional mode to any desired pitch trim angle at a rate established by the flow restrictors and bypass valves for that mode of flight.

In the VTOL flight mode, the stabilizer is automatically maintained at 20° leading edge up, through a VTOL range of $-5^{\circ}B_V$ to $+30^{\circ}B_V$. Between 30° and $45^{\circ}B_V$, it may be trimmed by the pilot at VTOL trim rates to establish longitudinal trim prior to conversion to CTOL. During conversion to CTOL, the stabilizer is automatically programmed at its maximum rate to -5° leading edge down. Subsequent to conversion, it may be trimmed by the pilot to the desired trim angles at the established CTOL trim rate.

In conversion from CTOL to VTOL, the stabilizer is automatically programmed to $+10^{\circ}$ leading edge up. At $30^{\circ}B_V$, it is further automatically programmed to 20° leading edge up, where it remains in VTOL mode, as mentioned above.

MAINTAINABILITY HISTORY AND RECOMMENDATIONS

Failure Rate

The cumulative average system failure rate should be based on the total number of operating hours, which would include both flight time and ground operating time. A more realistic failure rate is derived by combining the two times, since ground operations conducted when the aircraft was operating under tether conditions induced as much (if not more) stress and loads on the aircraft as did flight operations. The maintenance man-hours expended in correcting discrepancies are shown in Table XI.

It must not be overlooked that the aircraft was undergoing functional tests in the hangar by using the hydraulic ground cart to supply pressure to both hydraulic systems. This requirement also added to the accumulative wear-out factors and amounted to 397:50 functional test hours on system No. 1 and 217:10 functional test hours on system No. 2; this time is not included in the total operating hours.

DESIRABLE FEATURES

1. Two completely independent hydraulic systems for primary flight controls, with each system pump being driven by a different engine.

TABLE XI. MAN-HOURS EXPENDED IN CORRECTING DISCREPANCIES IN HYDRAULIC SYSTEM

Part Name	Prepa-ration	Diag-nosis	Accessi-bility	Corrective Action	Reas-sembly	Check-out	Total Man-Hours	Remarks
Diverter Valve Hydraulic Flex Line	0.1	0.1	0.2	0.3	0.2	0.2	1.1	Line replaced
Forward Hydraulic Servo	0.2	0.2	0.2	11.0	0.2	0.2	12.0	Replaced
Restrictor, Hydraulic Horizontal Stabilizer	0.5	10.0/5.0*	0.5	0.7	0.5	0.5	12.7	Replaced - slow-speed trim rate was decreased by 35 seconds
Disconnect Hydraulic O-Ring	0.2	0.1	0.5	0.6	0.5	0.2	2.1	Replaced because of wear

*If 2 numbers are divided by a slash, the first number represents the man-hours; the second, the clock hours.

2. Dual-input (electrical and mechanical) servo valve on fan-mode control surface hydraulic actuator.

3. Limited authority SAS control.

4. Bootstrap reservoir to provide positive pressure on pump suction without use of bleed air pressure regulators and associated components which would otherwise be required.

5. Self-centering, three-position, four-way, solenoid-operated valves.

UNDESIRABLE FEATURES

1. Secondary subsystems on same system as pitch axis SAS control valve.

2. Screened restrictors.

RECOMMENDATIONS AND SUGGESTIONS FOR UNEVALUATED IMPROVEMENTS

Accessibility

Provide larger access opening to the hydraulic compartment.

Leakage

Eliminate the swivel blocks in the hydraulic lines at the trailing edge area of the wing and forward of the flaps. Leakage has been a problem in these areas.

Hydraulic Brake System

Add parking brakes and improve the hydraulic accumulator system. The braking action on the present aircraft is about 50 percent of what is required.

Hydraulic Hose Fittings

To facilitate the connection of fittings, design more space between connection points on the aircraft where the external hydraulic hoses from the hydraulic ground cart are connected to the aircraft.

Conversion

Design the hydraulic system so that any combination of systems would complete conversion. At present, during conversion, hydraulic system No. 1 must be supplied by the primary electrical system; or if hydraulic system No. 2 is used, it must be supplied by the standby electrical system.

Operation of Main Landing Gear

Design the main landing gear so that it can be operated by both hydraulic systems and so that the priority valve can be a simple shuttle valve. At present, the main landing gear is operated by the primary system only, except for the emergency extension; in the latter case, stored air is used and the priority valve must work correctly.

Wing-Fan Doors

Add an electrically operated hydraulic pump to the hydraulic system. At present, the wing-fan butterfly doors have to be manually opened during inspection and maintenance.

Flight Control System

Eliminate the possibility of damage to the flight control system that may be caused by movement of the flight control stick or flight control surfaces when the hydraulic pressure is removed. Bottoming the actuators is very easy and can induce heavy loads in the mechanical mixer and also can cause out-of-rig problems.

Mixer Push-Pull Rods

Eliminate the possibility of damaging the mechanical mixer push-pull rods. In the present configuration, force loads are applied when the wing-flap control switch is repositioned with electrical power on the aircraft without hydraulic pressure's being available to the systems.

STABILITY AUGMENTATION SYSTEM (SAS)

SYSTEM CONFIGURATION AND OPERATION

General Description

The SAS is installed in the XV-5A to provide three-axis stabilization during fan-mode flight (see Figures 53 and 54). The system consists of a gyro package and an amplifier package, each of which consists of dual channels, which are referred to as primary and standby. The two channels are identical with the exception of the gain controls.

Gain Control Panel

The primary channel gain controls are located on the cockpit instrument panel and therefore are readily accessible to the pilot for system evaluation. The standby channel gain controls are located in the amplifier and are adjustable only on the ground. In addition to gain controls, the pilot gain control panel contains switches that turn off individual primary channel axes. Transfer from one channel to the other is accomplished on command by the pilot through the transfer switch located on the control stick; in the event of loss of 28-volt dc primary channel power, transfer from primary to standby is automatic. During normal flight operations, the primary channel is used; the standby channel is used for emergencies only.

Rate Gyro Package

Aircraft disturbances are sensed by a rate gyro, which has an in-phase or out-of-phase, 400-cps voltage output that is proportional to aircraft angular rates. To compensate for disturbances, this signal is amplified, demodulated, and fed to the hydraulic actuators which position the wing-fan exit louvers and/or pitch-fan doors. These actuators are the same units which respond to the pilot's stick and rudder pedal inputs through the control linkage. The stabilization signals are wired directly to the first stage of the actuator servo valves, where they are summed with the mechanical inputs.

The gyro packages are mounted in the cockpit under the pilot's seat. Each package contains three individual subminiature gyros along with the circuitry required for operation from aircraft power. The gyros have a full-scale range of $\pm 30^{\circ}$ per second.

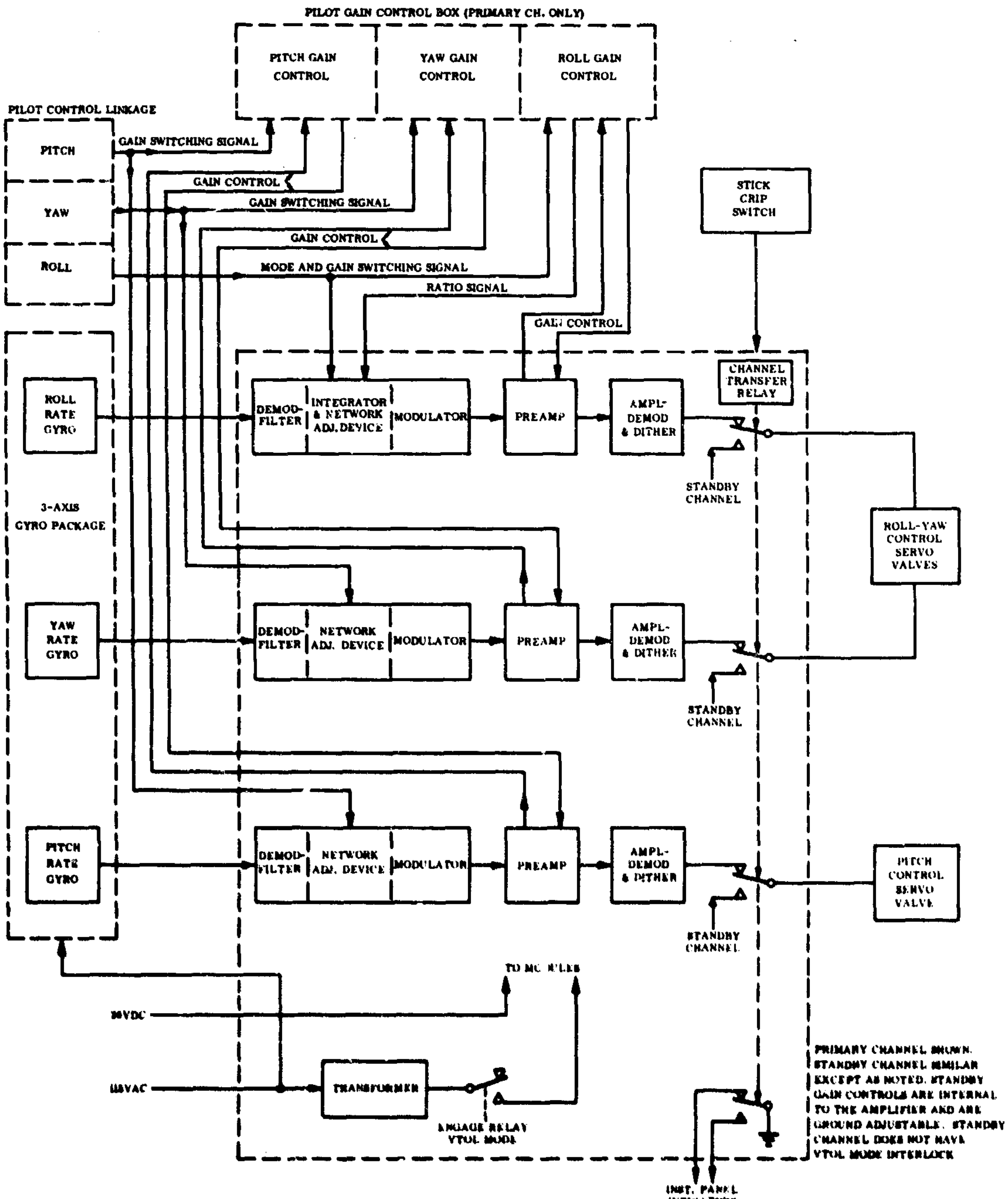


Figure 53. Automatic Stabilization System Block Diagram.

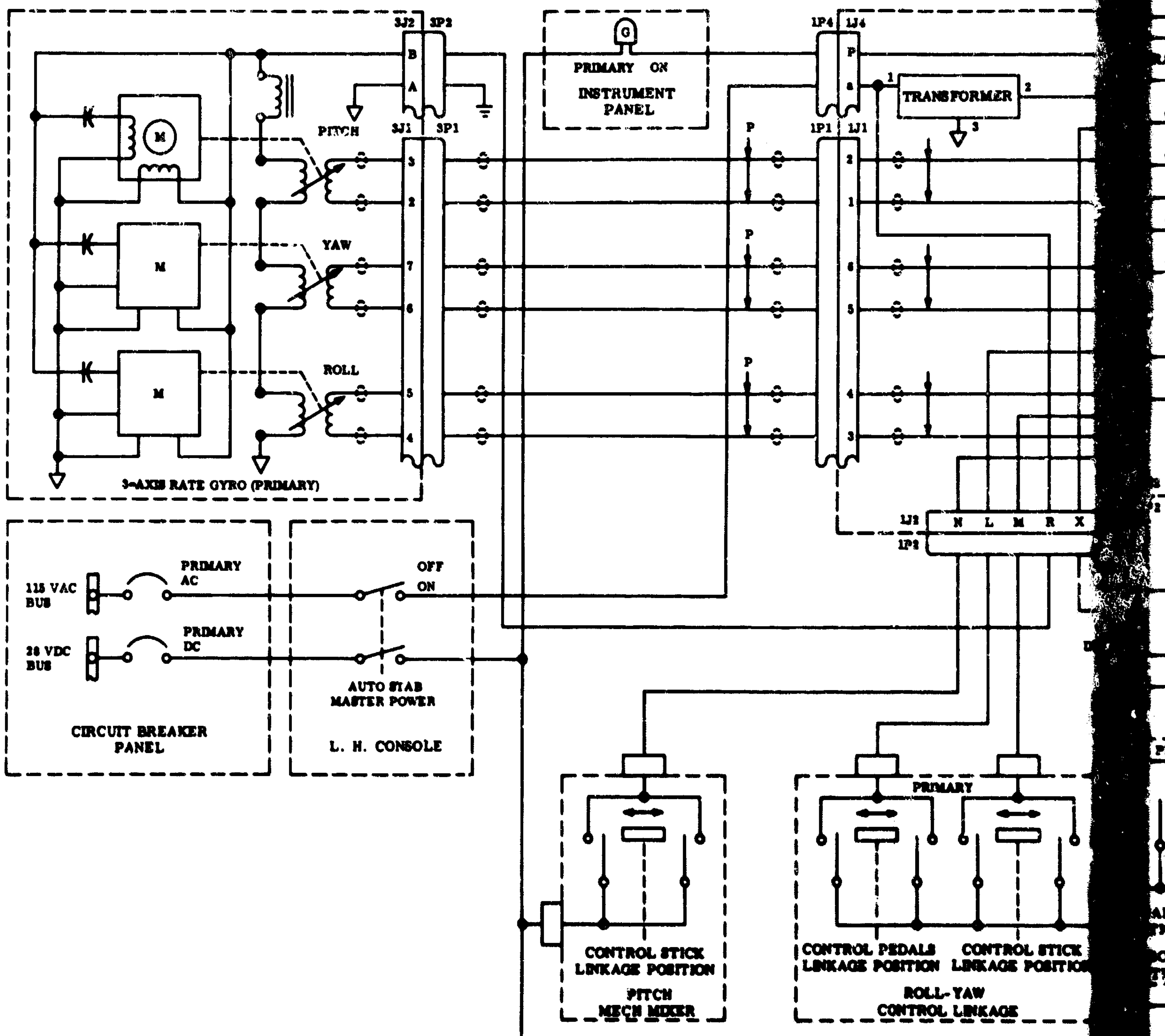
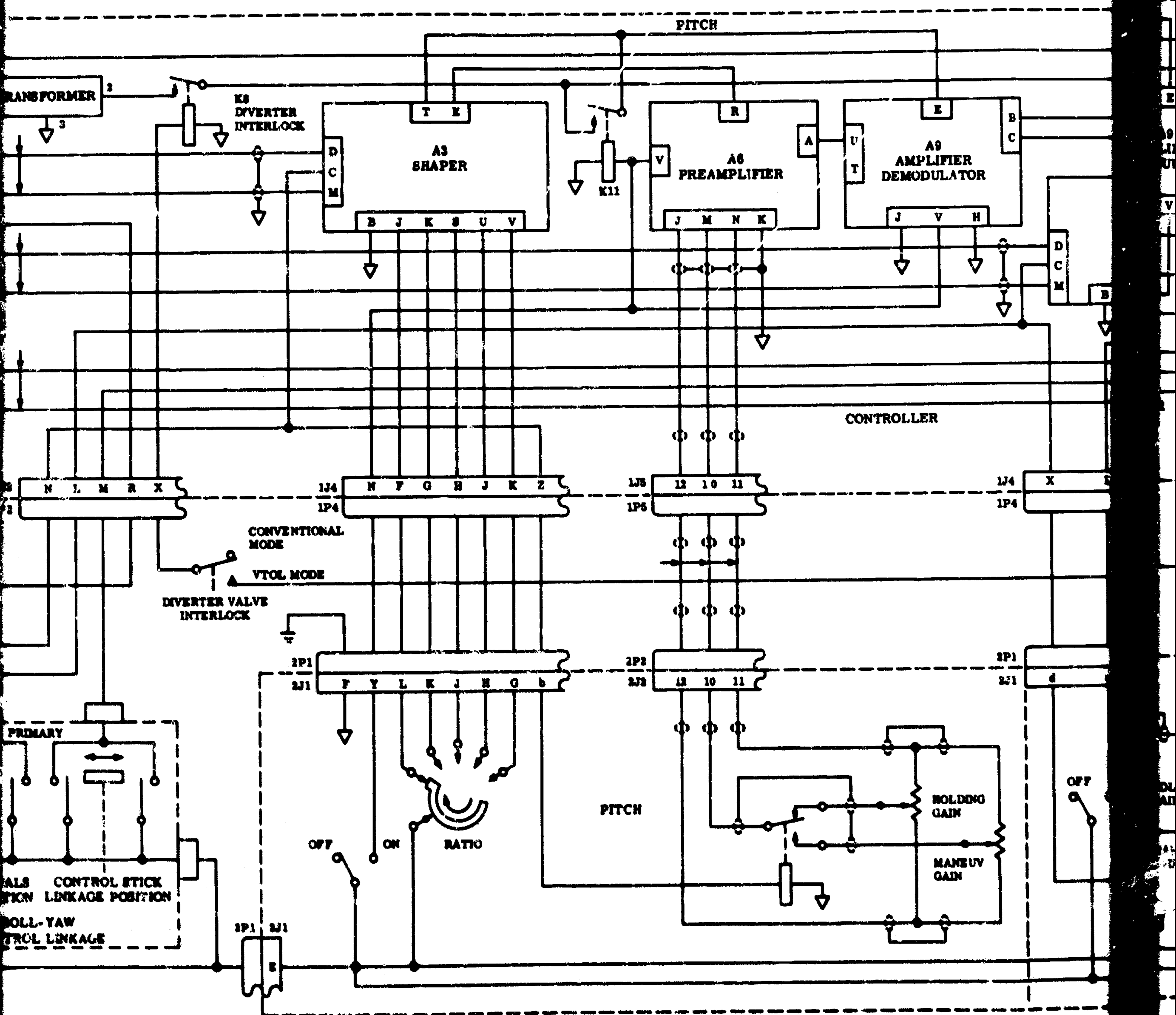
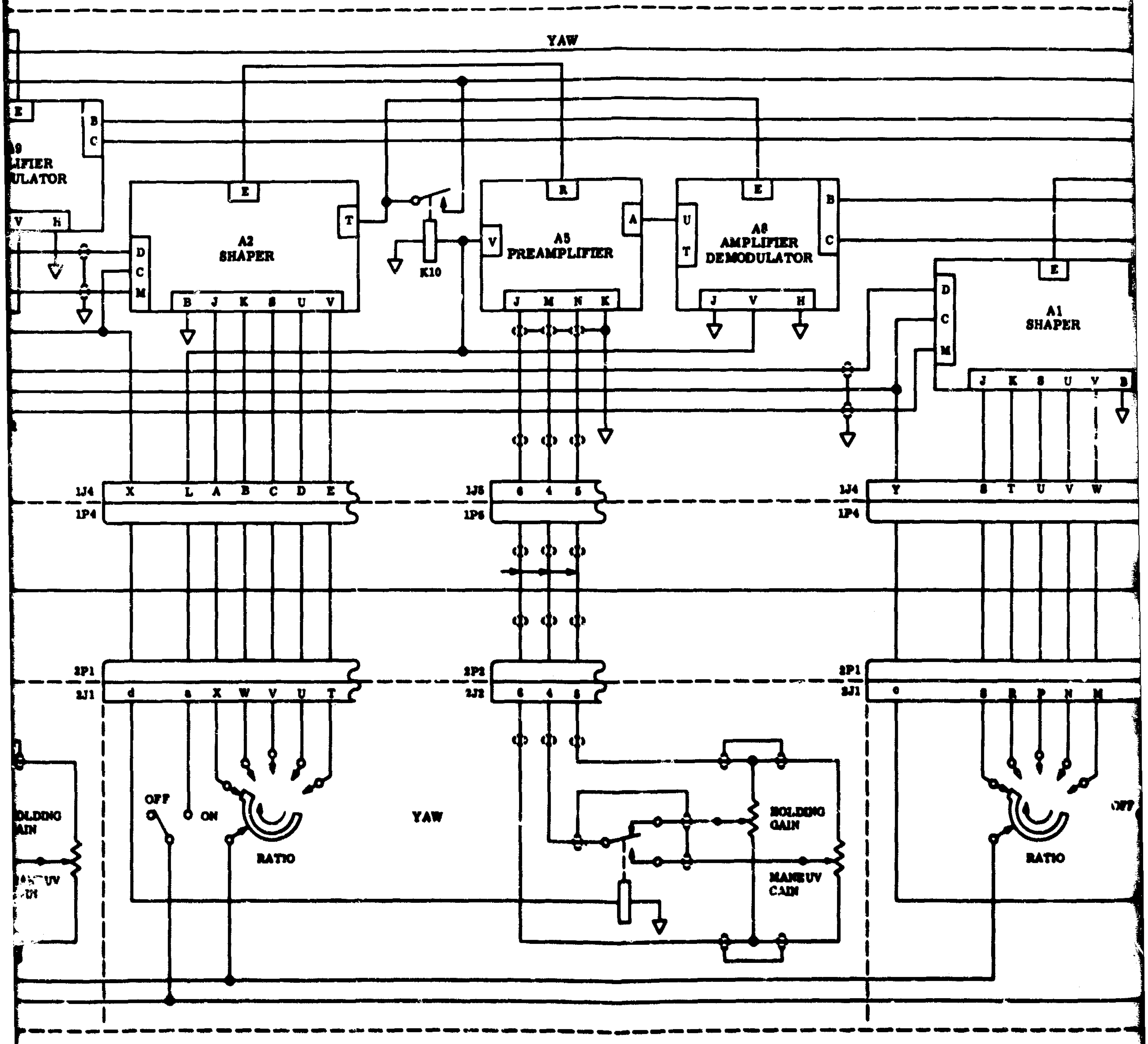


Figure 54. Automatic Stabilization System Schematic Diagram (Sheet 1 of 2).

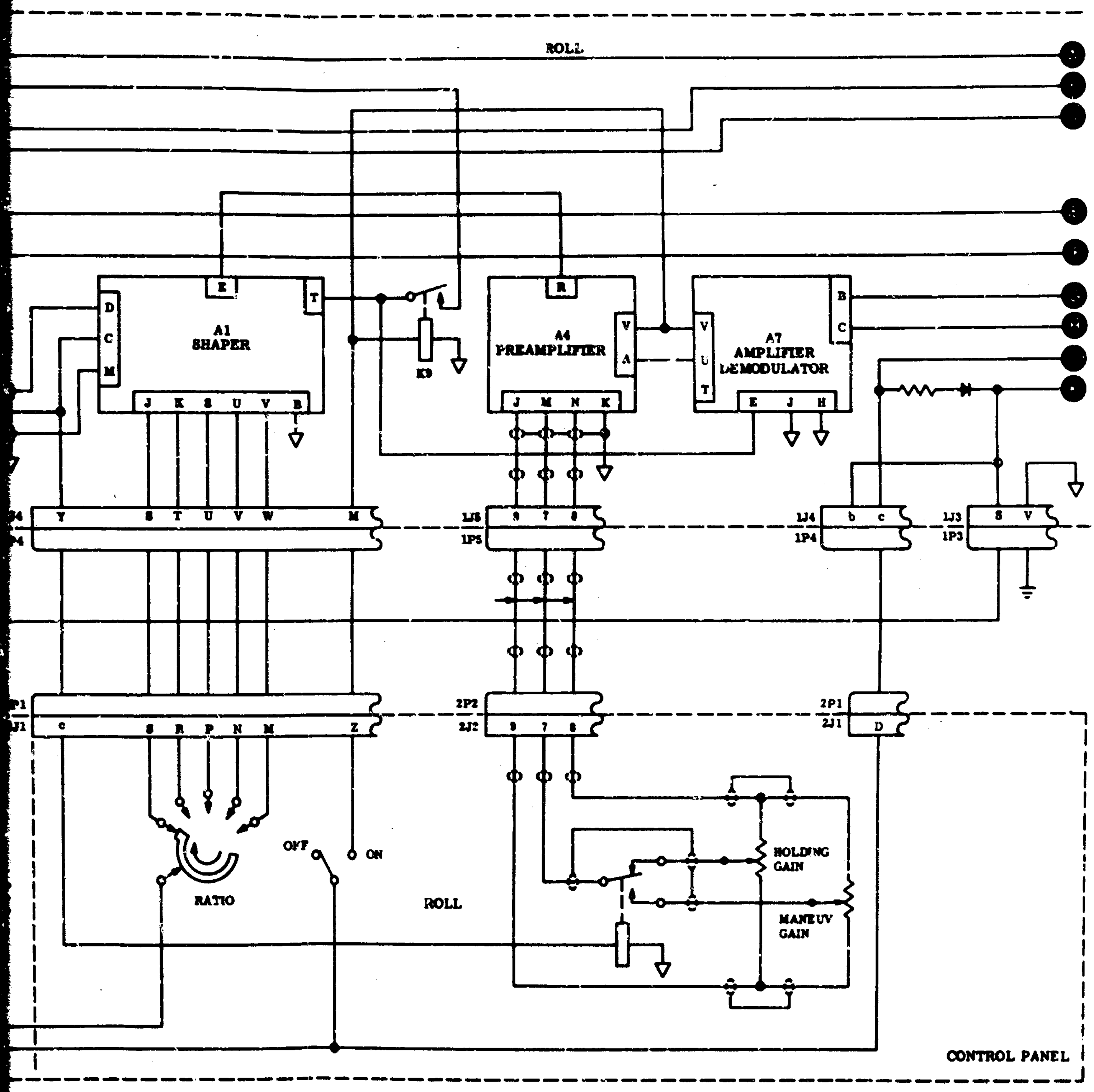

PITCH
RANSFORMER
K8
DIVERTER
INTERLOCK
A3
SHAPER
K11
A6
PREAMPLIFIER
A9
AMPLIFIER
DEMODULATOR
CONTROLLER
1J4
1P4
1J5
1P5
CONVENTIONAL
MODE
VTOL MODE
DIVERTER VALVE
INTERLOCK
2P1
2J1
2P2
2J2
PRIMARY
CONTROL STICK
LINKAGE POSITION
OLL-YAW
PITCH
OFF
ON
RATIO
HOLDING
GAIN
MANEUV
GAIN
OFF

B



YAW
A2
SHAPER
A5
PREAMPLIFIER
A8
AMPLIFIER
DEMODULATOR
A1
SHAPER
K10
1J4
1P4
1J5
1P5
2P1
2J1
2P2
2J2
OFF
ON
RATIO
YAW
HOLDING
GAIN

iC



ROLL
A1
SHAPER
K9
A4
PREAMPLIFIER
A7
AMPLIFIER
DEMODULATOR
RATIO
OFF
ON
ROLL
HOLDING
GAIN
MANEUV
GAIN
CONTROL PANEL

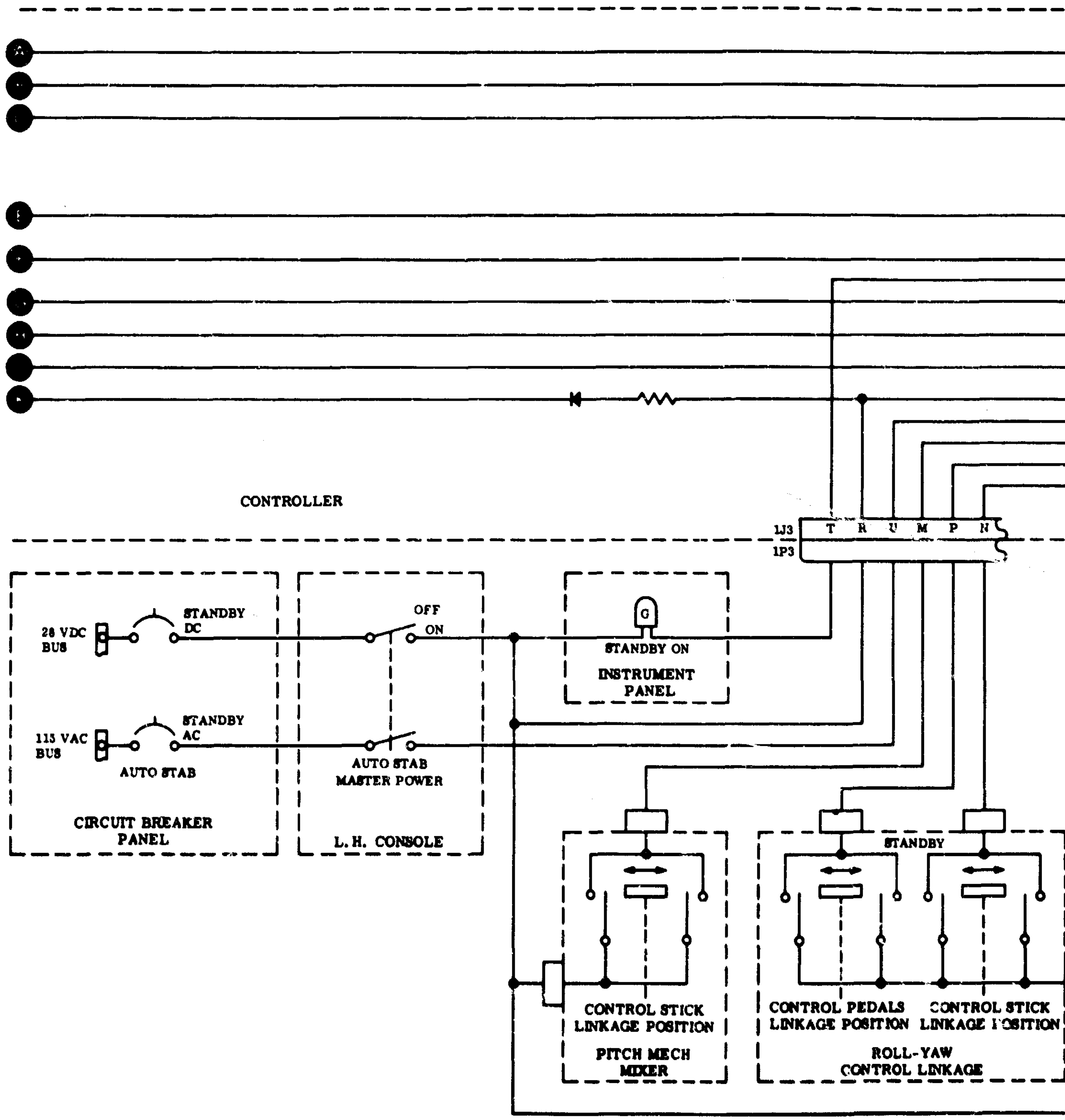
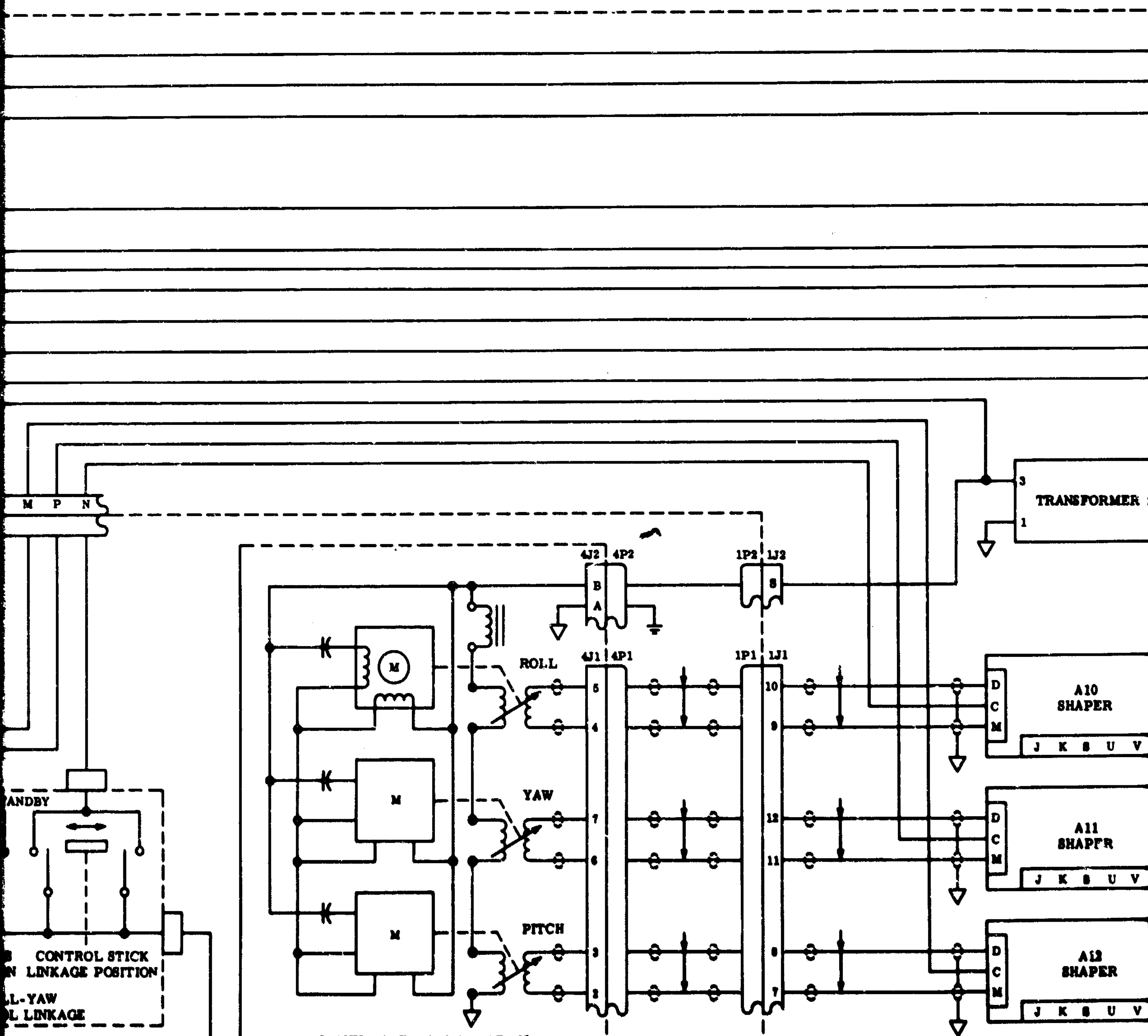
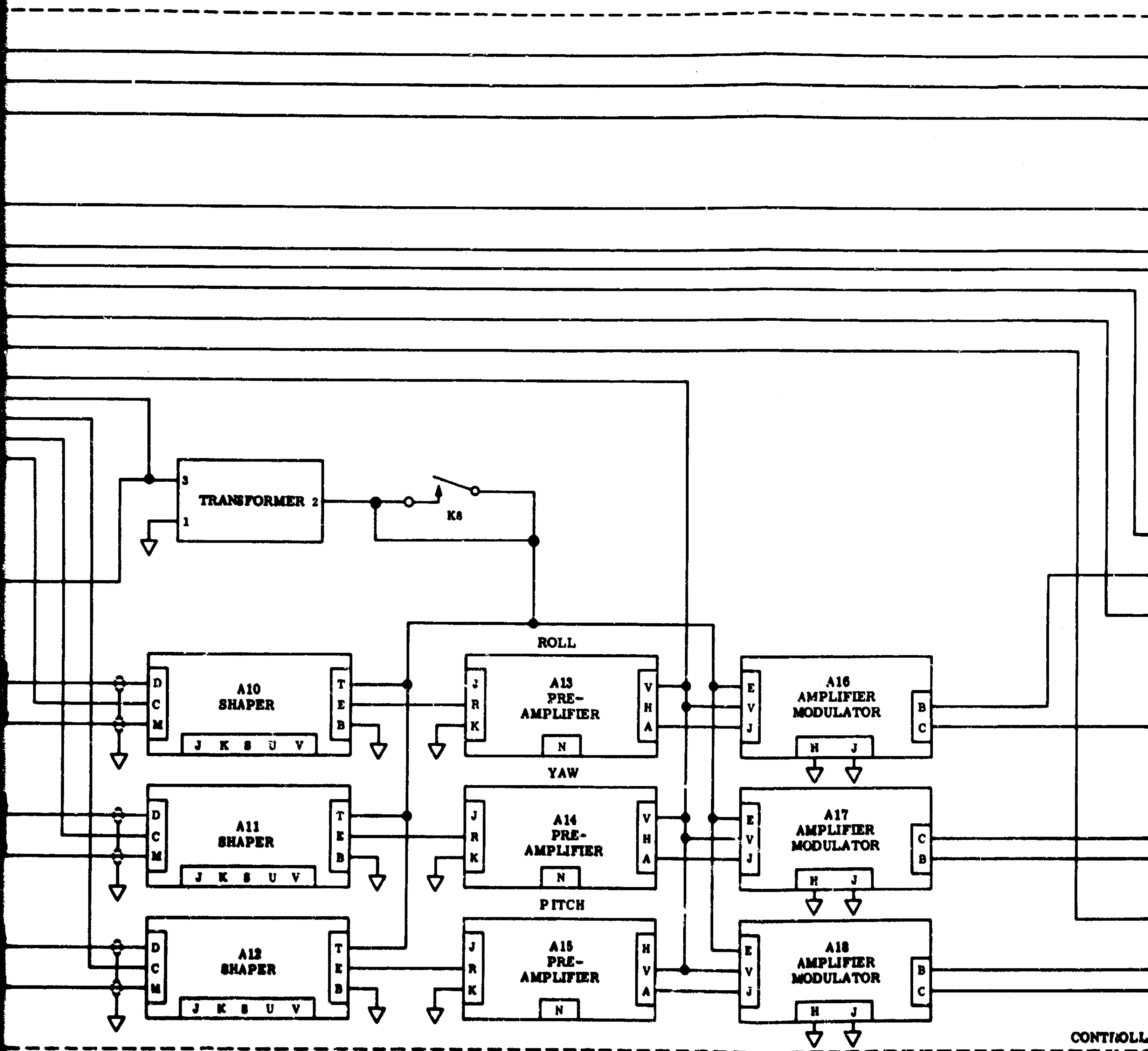


Figure 54. Automatic Stabilization System Schematic Diagram (Sheet 2 of 2).

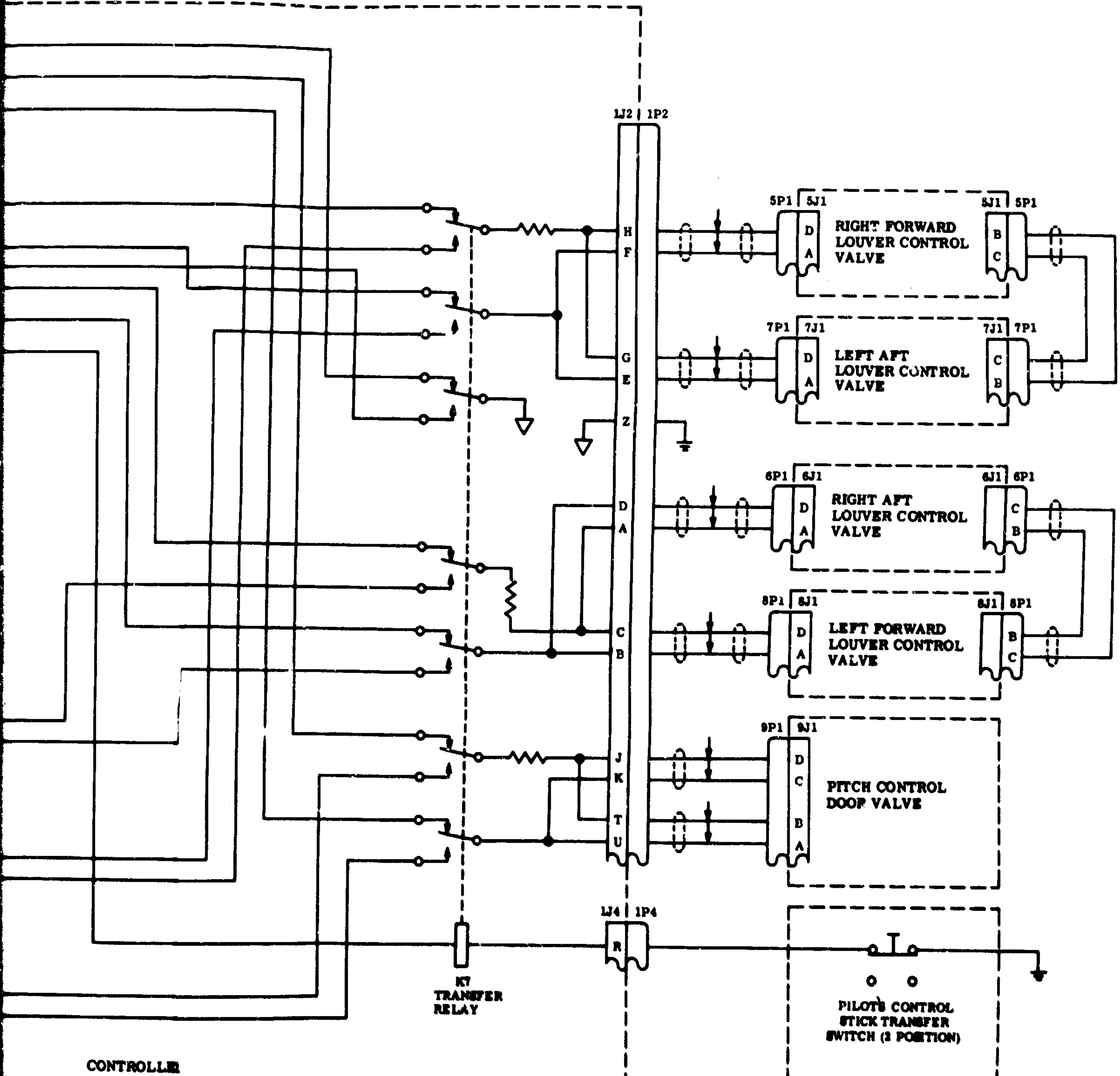

M P N
TRANSFORMER 2
3
1
4J2 4P2
1P2 1J2
B
A
8
4J1 4P1
1P1 1J1
ROLL
YAW
PITCH
M
A10
SHAPER
A11
SHAPFR
A12
SHAPER
D
C
M
J K 8 U V
ANDBY
CONTROL STICK
LINKAGE POSITION
L-YAW
L LINKAGE
3-AXIS RATE GYRO (STANDBY)

B



TRANSFORMER
K8
ROLL
YAW
PITCH
A10 SHAPER
A11 SHAPER
A12 SHAPER
A13 PRE-AMPLIFIER
A14 PRE-AMPLIFIER
A15 PRE-AMPLIFIER
A16 AMPLIFIER MODULATOR
A17 AMPLIFIER MODULATOR
A18 AMPLIFIER MODULATOR

C



1J2
1P2
5P1
5J1
RIGHT FORWARD LOUVER CONTROL VALVE
7P1
7J1
LEFT AFT LOUVER CONTROL VALVE
6P1
6J1
RIGHT AFT LOUVER CONTROL VALVE
8P1
8J1
LEFT FORWARD LOUVER CONTROL VALVE
9P1
9J1
PITCH CONTROL DOOR VALVE
1J4
1P4
K7 TRANSFER RELAY
PILOTS CONTROL STICK TRANSFER SWITCH (2 POSITION)
CONTROLLER

Amplifier Package

The amplifier package contains both the primary and the standby channel electronics and is located in the electrical equipment compartment. Solid-state circuitry is used throughout. Amplifiers are of modular construction. Each axis consists of three plug-in boards: the shaper board, the preamplifier board, and the amplifier-demodulator board. Of the nine boards in each channel, only two are not directly interchangeable between axes, but only a minor difference exists in these two. The amplifiers are designed to provide ±8 milliamperes to the actuator torque motor coils in the parallel configuration of the pitch-fan door actuator or bridge configuration of the louver actuators.

Maneuvering Mode Switch Packages

With the stick and rudder centered, or very nearly centered, the three axes of the system operate in the holding mode. For larger control displacements, ±1 inch from center, switches located on the control linkage change the individual axes to the maneuvering mode. In the holding mode, the axis gain is higher than it is in the maneuvering mode. In addition, in the holding mode, the roll signal from the gyro is integrated to obtain a quasi-attitude signal which is combined with the rate signal in a preselected schedule of ratio adjustments.

Control of Hydraulic Power

The fan-mode primary control system hydraulic actuator servo valves contain a single first stage where the electrical (SAS) and mechanical (pilot) inputs are summed. Hydraulically, this stage is operated by only one of the two hydraulic systems in order to preserve hydraulic system separation. During normal operation, hydraulic output of the first stage drives the second stage servo valve spool, which, in turn, controls the hydraulic power of both the No. 1 and the No. 2 systems to the tandem actuator. In the event that pressure is lost in the system supplying the valve first stage, mechanical (pilot) inputs automatically couple directly into the second-stage spool and continue to operate the actuator, with only minor changes in control surface response to pilot commands. However, when first-stage hydraulic pressure is lost, no actuator response to SAS command is possible. In order to prevent complete loss of SAS control due to the loss of one hydraulic system, the wing-fan louver actuator servo valve first stages are alternately powered by the No. 1 and No. 2 hydraulic systems; that is, the left-hand forward and right-hand aft by the No. 1 system, and the left-hand aft and right-hand forward by the No. 2 system (see Figure 50). Consequently, the loss of one hydraulic system results in only a 50-percent loss in roll and yaw axis stabilization. Since there is one tandem actuator (and only one servo valve) that operates the

nose-fan thrust modulator doors, loss of hydraulic system No. 1 removes all pitch axis stabilization electrically, but loss of hydraulic system No. 2 has no effect on pitch axis stabilization electrically.

Each servo valve first-stage torque motor consists of two coils. The wing-fan actuator coils are arranged in a bridge circuit to provide the proper summing of SAS roll and yaw commands. This bridge circuit is also especially configured such that no single-coil open- or short-circuit failure will cause loss of actuator SAS output. The first stage of the pitch control actuator servo valve torque motor also contains two coils. They are connected in parallel to provide the same single-coil failure protection.

Nominal authority of this system is 25 percent. That is, the actuator displacement due to maximum system input is one-quarter of the displacement due to maximum pilot input.

Electrical Input to Primary and Standby Channels

System operation requires both 28-volt dc and 115-volt ac, 400-cps power. Both the primary and the standby channels have individual circuit breakers that are connected to the 28-volt dc emergency bus and the 115-volt ac, 400-cps essential bus. System operation will continue even with the loss of both generators and one inverter. System ON condition is indicated by illumination of either the primary or the standby ON indicator light on the instrument panel; however, the primary channel is interlocked through the diverter valve, and neither louver nor pitch door motion will occur with the aircraft in jet-powered mode, even though the louvers and pitch doors are open. This is not the case for the standby channel; for this reason, the system should be kept in the primary channel for normal flight operations.

MAINTAINABILITY HISTORY AND RECOMMENDATIONS

Failure Rate

To date, no failures in the SAS have been reported.

Maneuvering Mode Switch Packages

Flight testing of the system has indicated that the attitude reference signal is not required in the pitch and yaw axes. The roll axis maneuvering mode switch packages are very difficult to adjust because of poor accessibility and the lack of rigid mounting structure. It is recommended that the roll

axis switch packages be relocated for better accessibility and that they be provided with a suitable mounting arrangement.

Self-Test Gyro

Preflight checkout of the SAS requires rocking the aircraft to verify that proper control surface directional responses are provided by the system. It has been suggested that a self-test gyro could be used to perform this check plus a system gain check, which is not possible with the existing configuration. A self-test gyro is available in the same package as is presently installed in the aircraft. It is recommended that such an instrument be evaluated for use on derivative aircraft.

DESIRABLE FEATURES

1. The SAS has two independent three-axis rate gyro packages; independent primary and standby amplifier circuits with manual pilot selection (by switch on control stick grip); and automatic transfer to standby if primary channel 28-volt dc power is lost.

2. Plumbing is routed from both hydraulic systems to the wing-fan servo valves. Thus, if power from one of the hydraulic systems is lost, the other system will continue to furnish power and thus the SAS will continue to operate.

3. The torque motor coil bridge circuit (wing-fan servo valve) and the torque motor coil parallel circuit (nose-fan servo valve) permit loss of one coil in either circuit without loss of SAS control commands.

UNDESIRABLE FEATURES

1. Roll axis maneuvering mode gain switches are difficult to adjust.

2. Control surface responses to SAS commands are difficult to check during ground preflight checks.

3. Control capability is not completely redundant in the hydraulic system, although SAS pitch axis commands are dually redundant.

COCKPIT GENERAL ARRANGEMENT AND SUBSYSTEMS DETAILS

COCKPIT CONFIGURATION AND SUBSYSTEMS

Cockpit Controls Installation

The cockpit is located in the forward fuselage section behind the nose fan. Side-by-side seating is provided for pilot and passenger. The pilot is located on the left side of the cockpit and has complete operational authority. The right-hand station is provisioned for either an observer or the flight data acquisition system. (The flight test data acquisition system was installed throughout the flight test program.) Access is gained through a manually operated canopy, hinged at the aft end.

All glass is of a nonsplintering type. Vision is at a maximum with minimum distortion, and glass areas have a luminous transmittance in excess of 70 percent. Precautions have been taken to reduce bothersome reflections; windows are of Plexiglas 55.

The general arrangement of the equipment in the cockpit is shown in Figure 55. All equipment mounted in the cockpit is installed to withstand ±40g fore-and-aft, ±30g vertical, and ±15g lateral crash conditions.

Instrument Panel Installation

The instrument panel contains the basic flight instruments (airspeed, altimeter, and magnetic compass), instruments required for special flying (including angle-of-attack indicator and sideslip angle indicator), and instruments normal to medium-performance jet aircraft (vertical speed, turn and bank, attitude, and acceleration). Other instruments included are position indicators that provide position readings for the louver vector angle, for flaps, for the thrust spoiler, and for trim in longitudinal, lateral, and yaw directions for both fan-mode and jet-mode flight. A master caution light and annunciator system is incorporated to provide subsystem malfunction and other hazardous condition information for the pilot. In addition, several normal-condition visual signal lights are provided. The mounting and types of power plant instruments are typical of power plant installations normally found in dual-engine jet aircraft. The instruments are mounted from top to bottom, with left-engine instruments on the left and right-engine instruments on the right. The power plant instruments comprise a tachometer and gages for exhaust gas temperature, fuel flow, fuel quantity, and oil pressure. The instruments have

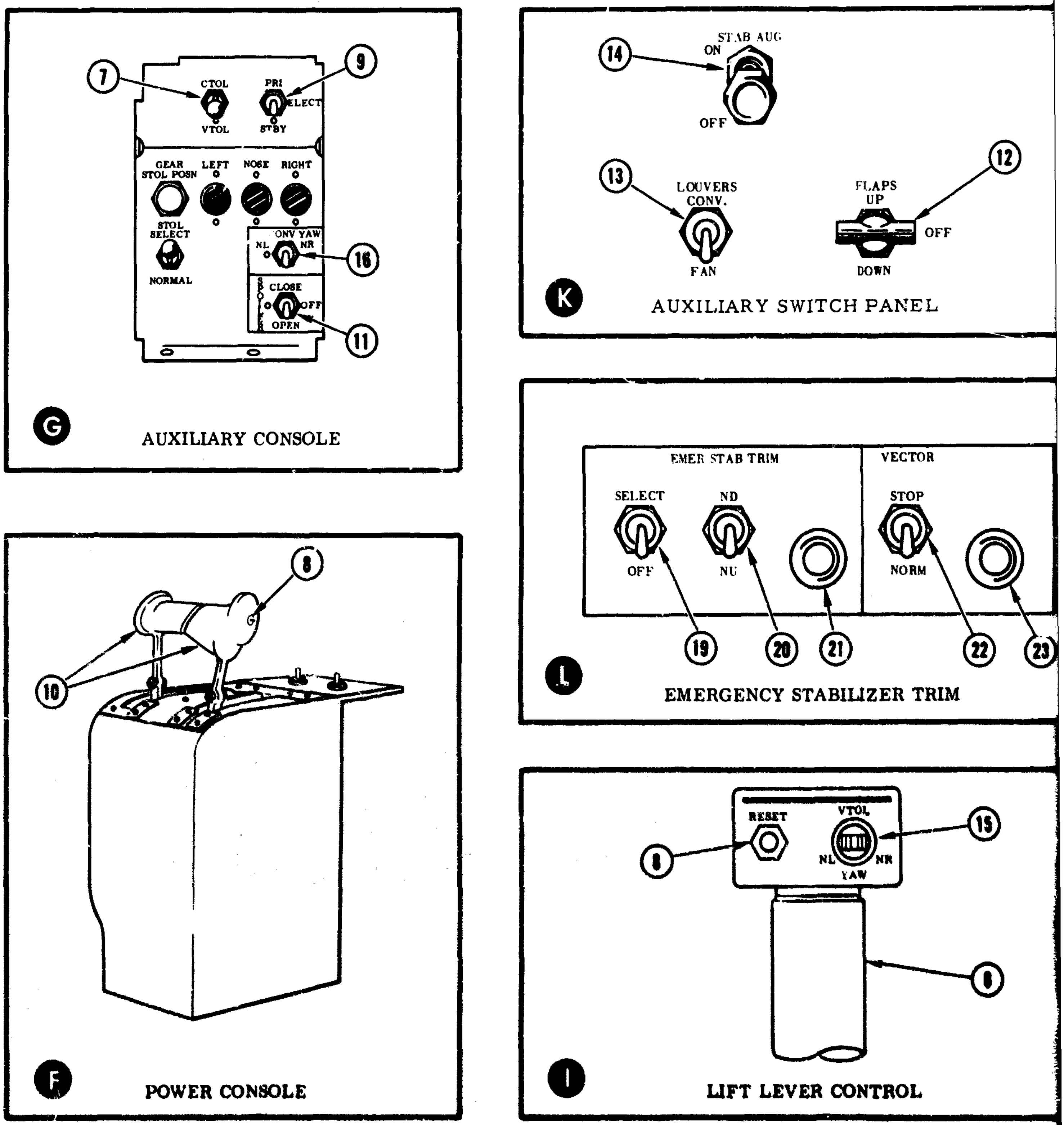
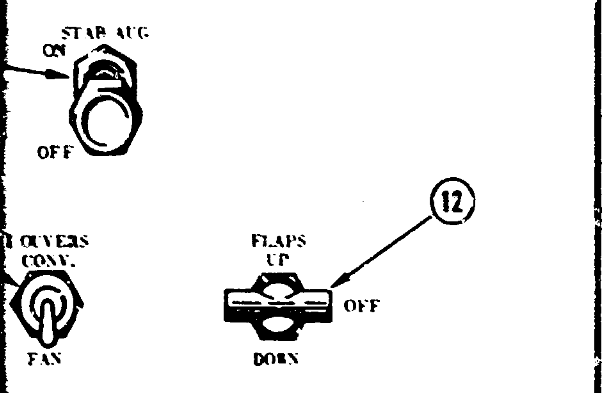
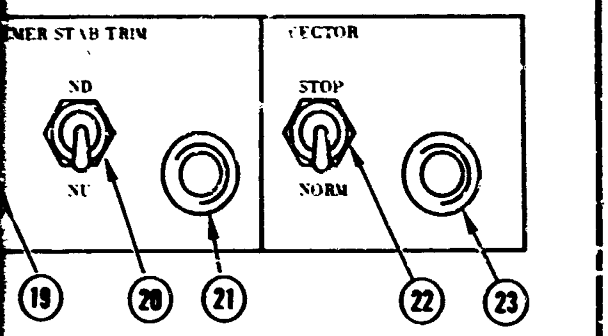


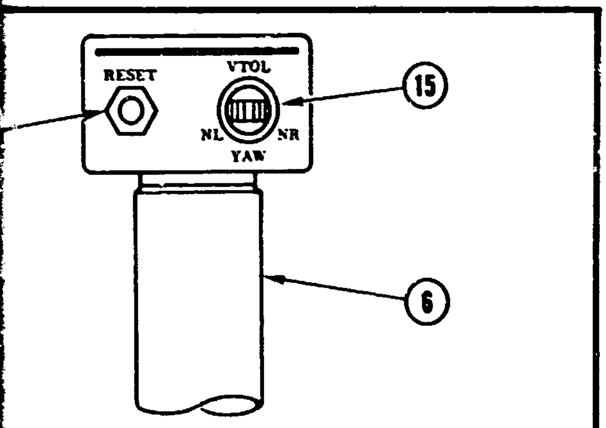
Figure 55. Crew Station Arrangement Diagram (Sheet 1 of 2).



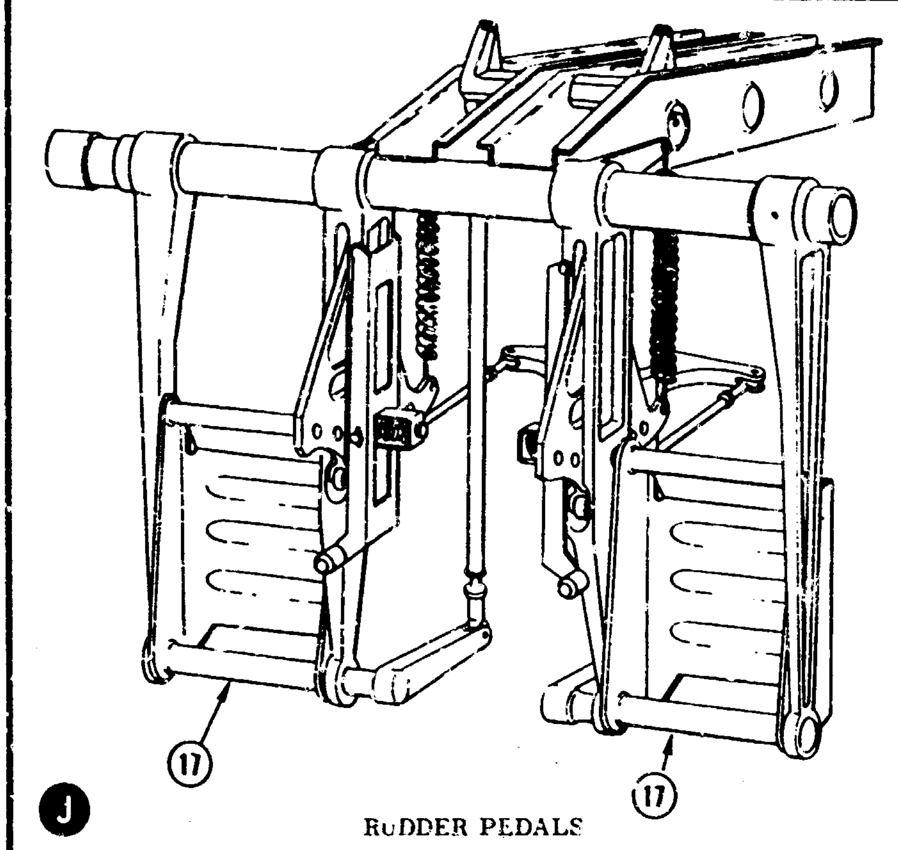
XILIARY SWITCH PANEL



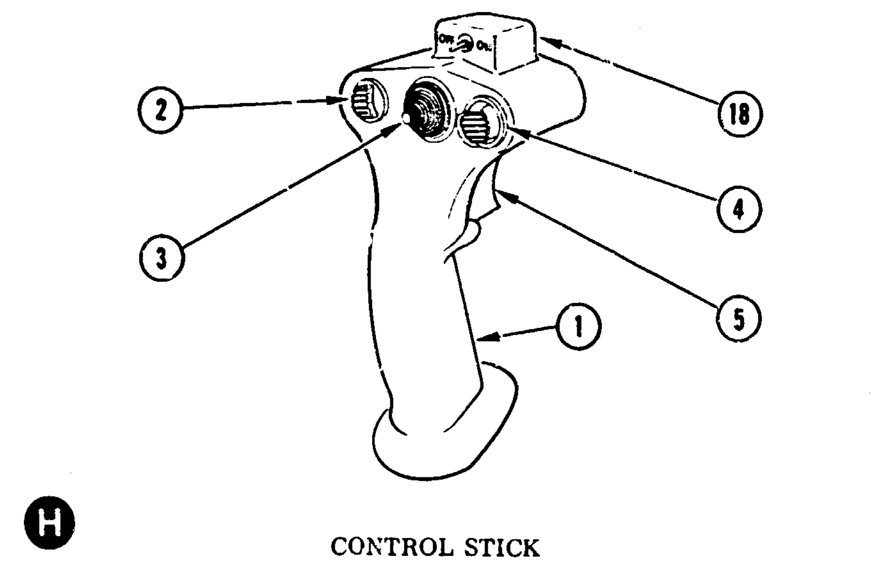
MERGENCY STABILIZER TRIM



LIFT LEVER CONTROL



RUDDER PEDALS



CONTROL STICK

of 2).

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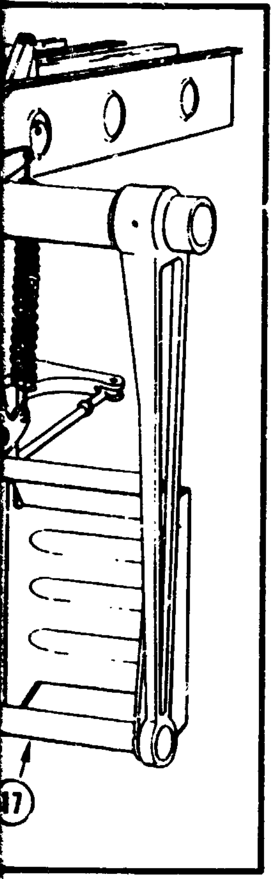
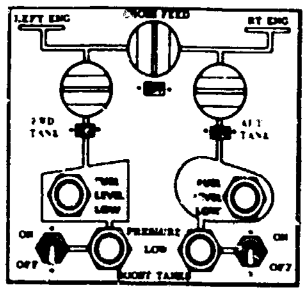
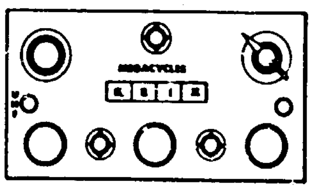
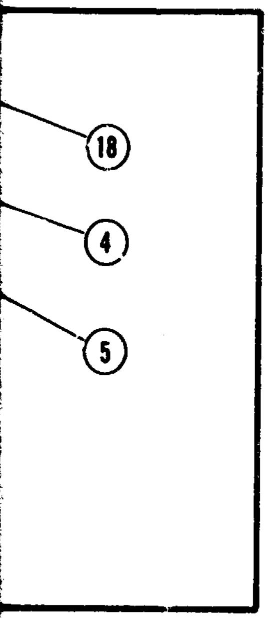


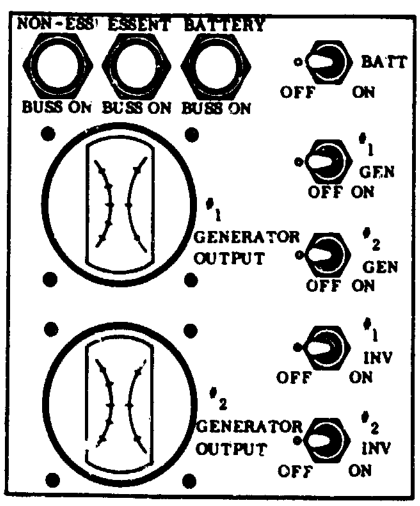


B FUEL CONTROL PANEL

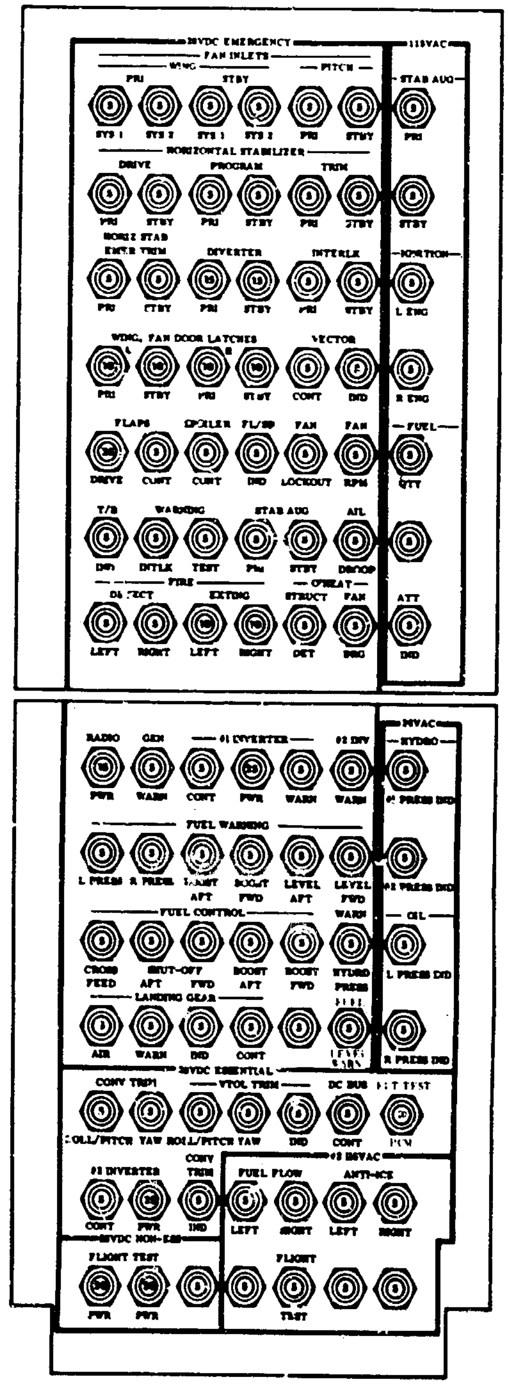


C UHF RADIO



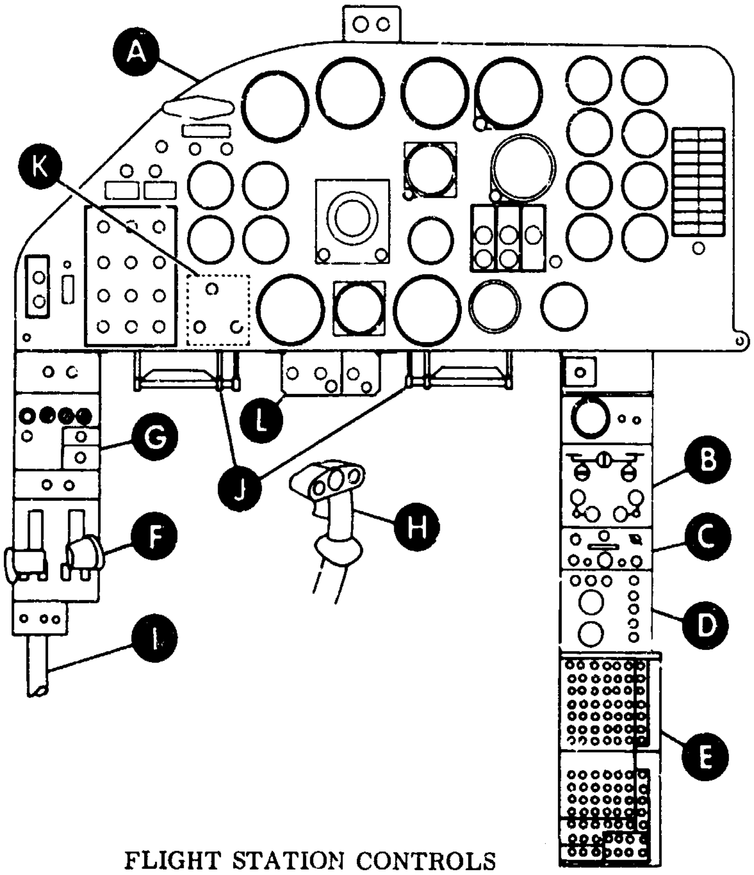


D ELECTRICAL PANEL



E CIRCUIT BREAKER PANEL

C



FLIGHT STATION CONTROLS

	CONTROL	FAN MODE	JET MODE
1	**Longitudinal Stick (Pitch)**	**Controls nose-fan thrust modulator doors**	**Positions elevator**
1	**Lateral Stick (Roll)**	**Controls differential wing-fan exit louver stagger**	**Positions ailerons**
2	**Exit Louver Collective Vector**	**Commands exit louver deflection**	**Inoperative**
3	**Roll-Pitch Trim**	**Adjusts centering devices for position of the stick for zero lateral and longitudinal forces at louver angles below 30°. Roll positions aileron tab and pitch positions horizontal stabilizer at louver angles above 30°.**	**Roll positions aileron tab and pitch positions horizontal stabilizer**
4	**Stability Augmentation System Primary-Standby Switch**	**Commands selection of the primary or standby stability augmentation systems**	**Operative as in fan**
5	**Microphone button**	**Operative as in jet mode**	**Rad! . transmit control**

	CONTROL	
6	Collective Lift (Altitude Up and and Down Motion)	Controls Coll stagger and n doors
6	Collective Power (Rotary Motion of Handle)	Controls gas ively
7	Mode Selector	Commands co
8	Power Reset (2 places)	Commands re erator power matic cutback speed
9	Electrical System Primary-Standby Switch	Operative as
10	Power Levers	Operative as
11	Thrust Spoiler Control	Operative as
12	Flap-Control Switch	Operative as closure comm
13	Louver Switch	Inoperative
14	Stability Augmentation System On-Off Control	Operative as
15	Hovering Yaw Trim	Adjust center the rudder p louver angle at louver an
16	Conventional Yaw Trim	Operative as
17	Rudder Pedals	Controls win vector
18	Instrumentation	To be used fo
19	Emergency Stabilizer Trim Switch	Operative as
20	NU-ND Switch	Operative as
21	Emergency Stabilizer Trim Light	Operative as
22	Vector Stop Switch	Stops vector
23	Vector Stop Light	Light on whe

Figure 55. Crew Station Arrangement Diagram (Sheet 2 of 2).

	FAN MODE	JET MODE
Up and	Controls Collective wing-fan exit louver stagger and nose-fan thrust modulator doors	Inoperative
Motion	Controls gas generator speed collectively	Operative as in fan mode
	Commands conversion to jet mode	Commands conversion to fan mode
	Commands return to original gas generator power setting following automatic cutback as a result of fan overspeed	Inoperative
ry-	Operative as in jet mode	Commands selection of the primary or standby conversion electrical system
	Operative as in jet mode	Controls gas generator speeds independently
	Operative as in jet mode	Commands thrust spoiler position
	Operative as in jet mode with closure command function removed	Controls flap position. Down position provides simultaneous command of flap deflection, nose-fan inlet louvers thrust modulating doors, and wing-fan exit louvers if louver switch is in fan position
	Inoperative	In fan position, provides for open command of nose fan inlet louvers, thrust modulating doors and wing-fan exit louvers simultaneously with flap down command. In conventional position, above functions are locked out.
ystem	Operative as in jet mode	Commands stability augmentation system on or off. Note: Stability augmentation system has no jet mode control authority
	Adjust centering devices for position of the rudder pedals for zero force at louver angles below 30°. Deactivated at louver angles above 30°.	Inoperative
	Operative as in jet mode	Adjusts rudder trim tab
	Controls wing-fan exit louver differential vector	Positions rudder
	To be used for instrumentation purposes	Operative as in fan mode
im Switch	Operative as in jet mode	Selects stabilizer emergency trim control
	Operative as in jet mode	Light on indicates emergency trim selected
im Light	Operative as in jet mode	Use for nose up or down trim when emergency stabilizer trim selected.
	Stops vector actuator runaway	Inoperative
	Light on when switch in stop position	Inoperative

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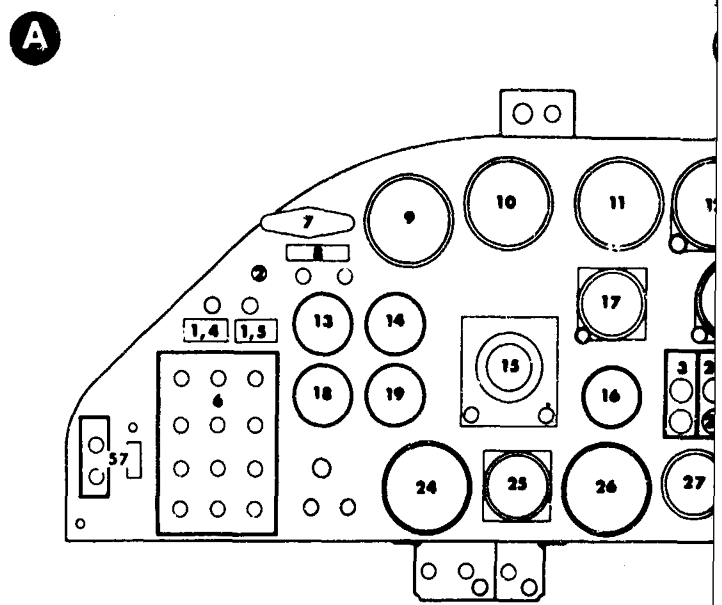

INSTRUMENT PANEL

1 EXTINGUISHING AGENT DISCHARGE SWITCH
2 EXTINGUISHING AGENT SELECTOR SWITCH
3 AUTO-STAB PRIMARY OR STANDBY INDICATOR
4 FIRE WARNING INDICATOR
5 FIRE WARNING INDICATOR
6 AUTO-STAB CONTROL PANEL
7 DRAG CHUTE LEVEL
8 MASTER CAUTION INDICATOR
9 ANGLE-OF-ATTACK INDICATOR
10 AIRSPEED INDICATOR
11 MACH METER
12 VERTICAL SPEED INDICATOR
13 FLAP AND THRUST SPOILER POSITION INDICATOR
14 VECTOR ANGLE INDICATOR
15 ATTITUDE INDICATOR
16 SIDE-SLIP INDICATOR
17 "G" ACCELEROMETER
18 NOSE-FAN TACHOMETER
19 DUAL-WING-FAN TACHOMETER
20 FAN DOOR "LOCKED" INDICATOR LIGHT
21 FAN DOOR "UNLOCKED" INDICATOR LIGHT
22 DIVERTER VALVE INDICATOR (FAN)
23 ALTITUDE INDICATOR
24 VTOL TRIM INDICATOR
25 TURN AND BANK INDICATOR
26 CTOL TRIM INDICATOR
27 CLOCK
28 LEFT ENGINE TACHOMETER
29 RIGHT ENGINE TACHOMETER

30 LEFT ENGINE EXH
31 RIGHT ENGINE EXH
32 LEFT ENGINE FUE
33 RIGHT ENGINE FUE
34 FUEL QUANTITY G
35 DUAL OIL PRESSUR
36 HYDRAULIC PRESS
37 ANNUNCIATOR PAN
38 FAN OVERSPEED W
39 FAN BEARING OVE
40 GENERATOR L.H.
41 GENERATOR R.H.
42 INVERTER NO. 1 O
43 INVERTER NO. 2 O
44 NO. 1 HYDRAULIC
45 NO. 2 HYDRAULIC
46 LANDING GEAR EM
47 STRUCTURE OVER
48 LEFT ENGINE LOW
49 RIGHT ENGINE LO
50 AFT LOW-BOOST-
51 FWD LOW-BOOST-
52 (NOT USED)
53 INTERLOCKS "NO C
54 PITCH-FAN FRAM
55 FUEL LEVEL LOW
56 MAGNETIC COMPA
57 LANDING GEAR CO

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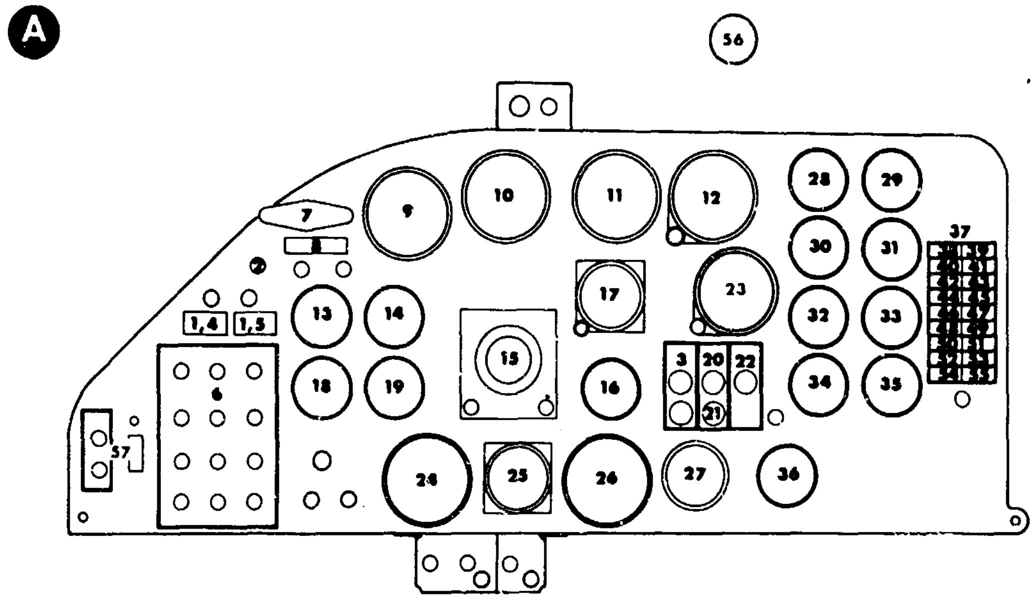


INSTRUMENT PANEL

1 EXTINGUISHING AGENT DISCHARGE SWITCH
2 EXTINGUISHING AGENT SELECTOR SWITCH
3 AUTO-STAB PRIMARY OR STANDBY INDICATOR
4 FIRE WARNING INDICATOR
5 FIRE WARNING INDICATOR
6 AUTO-STAB CONTROL PANEL
7 DRAG CHUTE LEVEL
8 MASTER CAUTION INDICATOR
9 ANGLE-OF-ATTACK INDICATOR
10 AIRSPEED INDICATOR
11 MACH METER
12 VERTICAL SPEED INDICATOR
13 FLAP AND THRUST SPOILER POSITION INDICATOR
14 VECTOR ANGLE INDICATOR
15 ATTITUDE INDICATOR
16 SIDE-SLIP INDICATOR
17 "G" ACCELEROMETER
18 NOSE-FAN TACHOMETER
19 DUAL-WING-FAN TACHOMETER
20 FAN DOOR "LOCKED" INDICATOR LIGHT
21 FAN DOOR "UNLOCKED" INDICATOR LIGHT
22 DIVERTER VALVE INDICATOR (FAN)
23 ALTITUDE INDICATOR
24 VTOL TRIM INDICATOR
25 TURN AND BANK INDICATOR
26 CTOL TRIM INDICATOR
27 CLOCK
28 LEFT ENGINE TACHOMETER
29 RIGHT ENGINE TACHOMETER
30 LEFT ENGINE EXHAUST GAS TEMPERATURE GAGE
31 RIGHT ENGINE EXHAUST GAS TEMPERATURE GAGE
32 LEFT ENGINE FUEL FLOW INDICATOR
33 RIGHT ENGINE FUEL FLOW INDICATOR
34 FUEL QUANTITY GAGE
35 DUAL OIL PRESSURE GAGE
36 HYDRAULIC PRESSURE INDICATOR
37 ANNUNCIATOR PANEL
38 FAN OVERSPEED WARNING LIGHT
39 FAN BEARING OVERHEAT WARNING LIGHT
40 GENERATOR L. H. OFF WARNING LIGHT
41 GENERATOR R. H. OFF WARNING LIGHT
42 INVERTER NO. 1 OFF WARNING LIGHT
43 INVERTER NO. 2 OFF WARNING LIGHT
44 NO. 1 HYDRAULIC SYSTEM LOW-PRESSURE WARNING LIGHT
45 NO. 2 HYDRAULIC SYSTEM LOW-PRESSURE WARNING LIGHT
46 LANDING GEAR EMERGENCY LOW-AIR-PRESSURE WARNING LIGHT
47 STRUCTURE OVERHEAT WARNING LIGHT
48 LEFT ENGINE LOW-FUEL-PRESSURE WARNING LIGHT
49 RIGHT ENGINE LOW-FUEL-PRESSURE WARNING LIGHT
50 AFT LOW-BOOST-PRESSURE WARNING LIGHT
51 FWD LOW-BOOST-PRESSURE WARNING LIGHT
52 (NOT USED)
53 INTERLOCKS "NO GO" WARNING LIGHT
54 PITCH-FAN FRAME OVERHEAT
55 FUEL LEVEL LOW
56 MAGNETIC COMPASS
57 LANDING GEAR CONTROL

C

been qualified in other aircraft with equivalent or more stringent requirements. Power levels during fan flight are monitored by fan rpm indicators. Fan-cavity temperatures also are indicated.

Communication Equipment Installation

An AN/ARC 51X UHF communication radio is provided. The system consists of a radio set, which is installed in the electronics compartment; a control panel, which is mounted on the cockpit center console (see Figure 55); and an antenna, which is mounted on the lower forward fuselage. An AM-843A/AIC-10 audio frequency amplifier is also included; it provides impedance matching for the pilot's microphone and headset and an audio output to the flight test instrumentation tape recorder. The amplifier is mounted in the electronics compartment.

Canopy Installation

The canopy installation consists of the canopy and hinges, an operating handle and locking mechanism (including emergency external operating handles), support rods, and a safety lanyard. Canopy operation is manual. The locking mechanism provides three positions (open, vent, and closed); it is operated by a control handle located on the right-hand side of the center console. The safety lanyard is provided to permit taxiing with the canopy in the vent position. When the locking handle is in the open position, the canopy rests on the open locking hooks, in approximately the vent position. It may then be manually raised to the full-open position, where it is held by two support rods. The support rods are stowed along the cockpit sidewalls when not in use. Open hinges permit easy canopy removal for cockpit maintenance access and for emergency egress conditions. The canopy is designed for through-the-canopy ejection. The canopy can be unlocked and opened externally from either side of the aircraft with a single continuous motion by a person standing on the ground.

Cockpit Ventilation System

The cockpit is ventilated by outside air drawn into the cockpit through slots formed by the canopy closure; air is exhausted from the cockpit by engine-driven blowers (see Figure 7). Pilot controls to regulate cockpit ventilation are not provided. Anti-icing, moisture control, and pressurization are not provided, since they are not required for the intended mission of the aircraft.

Antispin/Drag Parachute System

An antispin/drag parachute system is included as a flight safety aid during flight envelope expansion operations. Selection of the parachute configuration to be installed is dictated by the specific flight test mission requirements. Parachute deployment is accomplished by pulling the control handle straight out (see item 7 in block A of Figure 55). Rotating the handle 90° clockwise and pulling again releases the parachute. The parachute is not coupled to the aircraft until the deployment handle is actuated. If the retaining cover blows off, the parachute is automatically jettisoned.

Pitot-Static System

The Pitot-static system consists of two boom-mounted Pitot heads, plumbing (with moisture drains), a manifold in the cockpit, hoses for instrument attachment, and a solenoid-controlled pneumatic valve to shut off the low-airspeed indicator during high-speed flight. A nose boom and a boom on the left wing tip support the two Pitot heads.

Pilot Oxygen System

A diluter/demand low-pressure gaseous oxygen system sufficient to supply up to 100-percent oxygen for the pilot is incorporated. The control valve and quantity gage are located on the cockpit aft bulkhead.

Pilot Ejection Seat

The pilot ejection seat is an LW-2 ejection system. It is capable of pilot recovery at zero speed and zero altitude. Provisions are also made for high-speed and/or high-altitude pilot recovery conditions.

MAINTAINABILITY HISTORY AND RECOMMENDATIONS

Failure Rate

Fourteen cockpit system failures were reported. Of these, six were reported during a flight or ground test. The remaining eight failures were reported during the various inspection and functional test procedures. The maintenance man-hours expended in correcting discrepancies are shown in Table XII.

Antispin/Drag Parachute

The highest failure rate for the cockpit subsystem occurred with the antispin/drag parachute. Uncommanded release of the retaining cap

Part Name	Prepa-ration	Diag-nosis	Accessi-bility	Corrective Action	Reas-sembly	Check-out	Total Man-Hours	Remarks
UHF Radio Assembly	0.1	0.1	0.1	0.2	0.1	0.1	0.7	Adjusted
Wing-Fan RPM Indicator	0.1	0.2	0.1	0.1	0.1	0.5	1.1	Replaced
Canopy Latch Assembly	0.1	0.1	0.5	3.6	0.5	0.2	5.0	Balls replaced
UHF Radio - ARC51X	0.2	0.2	2.0/1.0*	100.0/50.0	20.0/1.0	2.0/1.0	124.4	Replaced with ARC 34
UHF Radio - ARC 34	0 1	0.1	0.2	0.3	0.2	0.1	1.0	Loose connection tightened
Rate-of-Climb Indicator	0.1	0.2	0.1	1.5	0.1	0	2.0	Replaced
Wing-Fan RPM Indicator	0.2	0.4	0	0.2	0	0.2	1.0	Broken wire repaired

TABLE XII. MAN-HOURS EXPENDED IN CORRECTING DISCREPANCIES IN COCKPIT

*If 2 numbers are divided by a slash, the first number represents the man-hours; the second, the clock hours.

caused one fan-mode flight termination. In this case, the parachute was automatically jettisoned by the release mechanism while the parachute was still in the packing bag. This malfunction was attributed to differential thermal expansion between the steel push-pull cable and the aluminum sheath. After mid-June 1965, the parachute was removed and the retaining cap was taped in place.

Canopy Locking Mechanism

On several occasions, the cockpit canopy operating mechanism jammed in the locked position, and outside assistance was required to open the canopy. This problem was attributed to the extremely poor design of the canopy locking mechanism.

DESIRABLE FEATURES

1. The cockpit area is large, with ample space for the pilot.
2. The main in-flight controls and switches are within easy reach of the pilot.
3. The pilot's overall field of view is good (except for a few reports of restricted visibility downward and aft).

UNDESIRABLE FEATURES

1. Operating mechanism in antispin/drag parachute installation permits uncommanded parachute deployment.

2. High temperatures in the electronic compartment adversely affect radio reliability and maintenance requirements.

3. Cockpit lacks ventilation control.

RECOMMENDATIONS AND SUGGESTIONS FOR UNEVALUATED IMPROVEMENTS

Accessibility to Controls

All controls and switches, except the two minor controls described below, are within comfortable and easy reach of the pilot when his harness is locked. The oxygen-diluter demand control is located on the aft cockpit wall next to the ejection seat; although this location is adequate for test purposes, the control is awkward to reach and hard to see. The circuit-breaker panel is on the center flooring and cannot be reached by a pilot with short arms unless he releases the harness and leans sideways.

Modification of Fuel System

The existing fuel system incorporates a feed routing that is contrary to all recognized conventions and which, if left unchanged, will eventually lead to fuel mismanagement and to possible double-engine flameout.

All conventions and layout specifications call for numbering of engines, tanks, and other major items from left to right and from front to rear. This sequence is both natural and logical. The engines in the XV-5A are numbered correctly; that is, No. 1 engine is on the left and No. 2 engine is on the right. (If, as in some aircraft, these engines were longitudinally disposed, the front engine would be No. 1 and the rear engine would be No. 2.) The fuel tanks in the XV-5A are longitudinally disposed, and yet the rear tank (No. 2) feeds the left-hand engine (No. 1).

Fuel loadings in the XV-5A are critical for maintaining test center-of-gravity positions and require considerable fuel management. The unnatural and incorrect feeding sequence causes unnecessary hesitation and occasional confusion when rapid, accurate, and safe tank selection is required. If the fuel tanks are to be referred to as forward and aft, it is essential that they feed the No. 1 (left) and No. 2 (right) engines, respectively, to reduce to a minimum the chances of inadvertent shutoff of the

wrong tank. For example, when operating with the forward tank off, instructions from the tower to turn the forward tank on may lead to an instinctive operation of the left-hand selector, which would, in the case of the XV-5A, turn off the aft tank and result in a double flameout.

To resolve this deficiency, two simple modifications are possible. The tanks could be redesignated No. 1 for aft and No. 2 for forward, in which case no further mention should be made of the forward and aft designations. However, this solution is undesirable, since the center-of-gravity movement is better visualized if the forward and aft designations are used. Alternatively, the feed pipes from the two fuel filters, located beneath each engine, could be cross-routed to the opposite engine fuel control units, in which case the lettering on the fuel control panel should be altered to show FWD on the left-hand selector and AFT on the right-hand selector. This modification is strongly recommended, and it should not involve a weight penalty in excess of a few pounds.

Caution Light

The purpose of the caution light is to draw the pilot's attention to a warning on the annunciator panel. In the XV-5A, the following deficiencies have been observed:

1. The caution light is partially obscured by the drag-chute operating handle, so that it is difficult to see and awkward to cancel.

2. It is a steady rather than a flashing light, regardless of the nature of the warning on the annunciator panel.

3. It is effective only when the pilot is looking forward.

4. It is similar in appearance to the fire warning lights and is therefore susceptible to confusion in an emergency.

5. No audio warning is associated with its illumination.

An effective system should include the following characteristics:

1. Several lights should be used to attract attention, and they should be located so that at least one of them will be visible, regardless of pilot head position or field of view.

2. The lights should flash when triggered by low-priority warnings and should remain steady only for urgent warnings.

3. An audio signal should be synchronized with the light for urgent warnings.

4. The cancel button should be remote from any other buttons with which it could be confused.

Annunciator Panel

The annunciator panel, which is placed on the right-hand side of the instrument panel, is not angled toward the pilot; therefore, it is subject to sun glare and undue reflections. Occasionally, the pilot must lean to one side or the other to confirm lettering. The illumination of the segments is not bright enough to overcome bright sunlight. There is no distinction between urgent and precautionary warnings; for this reason, the degree of urgency is not immediately apparent to the pilot. There is no engine-failure warning indication.

Accessibility to Rear of Instrument Panel

Easier access to the back side of the main instrument panel should be provided.

Warning Lights for Fire and Overheating

The warning lights for fire and overheating might not be observed when the pilot's attention is directed outside the cockpit, such as during formation flying, transition to hover, hovering, landing, and taking off. The illumination of these lights should actuate attention-getting lights and audio warning signals. The discharge and selector switches detract from prompt and correct actuation and are subject to misuse in emergency conditions. When the decision to operate a fire bottle is made, no further decisions or reasoning should be required; a single pilot action should initiate the bottles.

LANDING GEAR SYSTEM

SYSTEM CONFIGURATION AND OPERATION

General Configuration

Figure 56 shows the general landing gear configuration, which is a nose-gear tricycle type. The forward retracting nose gear is of the conventional configuration. Nose-gear steering is not provided. Steering is by differential application of the main wheel brakes. The fuselage-mounted aft-retracting main gears are equipped with a mechanism that provides alternative positions of the main wheels. A tail bumper is provided to protect the aft end of the fuselage and the main gear doors in an extreme nose-up condition.

Main Landing Gear Installation

Each main landing gear consists of a vertically acting shock strut, supporting structure, and wheel and brake assembly. In the extended configuration, provisions are made for two different main wheel longitudinal locations. The gear pivots bodily about trunnions rigidly attached to the fuselage. The shock strut is an oleo pneumatic unit with a tapered metering pin and orifice. Shock strut inflation pressure is 170 psig. Shock strut stroke is 9.2 inches. The hydraulically actuated two-position mechanism directs the wheels forward and, when the gear is in the extended position, directs the wheels aft. The mechanism is designed to permit transfer of the wheels in either direction when the aircraft is on the ground. Transfer time is approximately 7 seconds. In the CTOL configuration, the mode-change transfer to the aft position is the first stage of gear retraction. When the gear is extended to the CTOL configuration, mode-change transfer to the forward position is the last stage of extension.

The forward position of the wheels is locked by an overcenter toggle mechanism. The aft position is locked by an internal lock in the mode-change actuator. Limit switches provide position and locking information to the cockpit indicator and warning system. The switches also form a part of the main gear sequencing system.

The gear is retracted aft into the fuselage belly by hydraulically actuated folding drag struts. Internal ball locks in the retract actuators lock the drag struts in the gear extended position. The gears are locked in the up

position by mechanical uplatches mounted on the wheel bay roof. The uplatches are hydraulically released prior to gear extension.

The main gear doors are of the double-folding clamshell type, hydraulically actuated through a mechanical linkage. The mechanism incorporates overcenter toggle linkages to lock the doors in the open and closed positions. The doors remain open when the gear is extended.

Limit switches on the gear retract actuators, on the uplatches, and on the door mechanism interlock the relative motion of gear and doors and provide information to the cockpit indicator and warning system. From the wheels-aft position, the retraction time from pilot command to doors closed is approximately 8 seconds.

The Type VII 20-by-4.4 tubeless tires have a 12-ply rating, rib tread, and an inflation pressure of 180 psig.

Nose Landing Gear Installation

The nose-gear assembly consists of a vertically acting shock strut with fork-type axle and single-wheel assembly. The gear assembly is attached to the fuselage, under the cockpit area, with trunnion fittings. The shock strut is an oleo pneumatic unit with a tapered metering pin and orifice. A recoil valve is provided to reduce fore and aft pitching. Shock strut inflation pressure is 160 psig. Shock strut stroke is 8.0 inches. Forward retraction is accomplished conventionally by a hydraulically actuated folding drag strut. An auxiliary jury strut provides locking in both extended and retracted positions. The doors are operated by direct mechanical linkages to the gear mechanism. Forward doors reclose after gear extension. Retraction and extension time amounts to approximately 4 seconds.

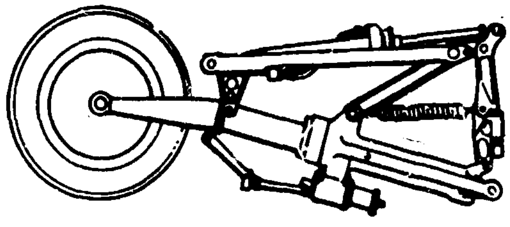
Position information for the cockpit indicators and warning system is provided by the jury strut lock switch and gear UP position switch. The nose wheel casters 70° each side of center for taxiing. A self-centering cam lines up the nose wheel from 30° either side of center prior to retraction. Disconnecting the torque links provides 360° castering for towing and ground handling.

The nose wheel is designed to caster under forward rolling motion only. In VTOL touchdowns, no rearward translation of the aircraft relative to the ground is permitted, since damage to the torque linkage or shimmy damper may result.

Nose-gear power steering is not provided. A vane-type shimmy damper is connected to the upper torque link for shimmy suppression. The damper

NOSE LANDING GEAR

1. NLG HYDRAULIC ACTUATO
2. NLG LOCKED SWITCH
3. JURY BRACE
4. SPRING
5. SHOCK STRUT
6. NLG WHEEL ASSEMBLY
7. SHIMMY DAMPER
8. GROUND LOCK PIN
9. TORQUE LINKS
10. UPPER DRAG BRACE
11. LOWER DRAG BRACE



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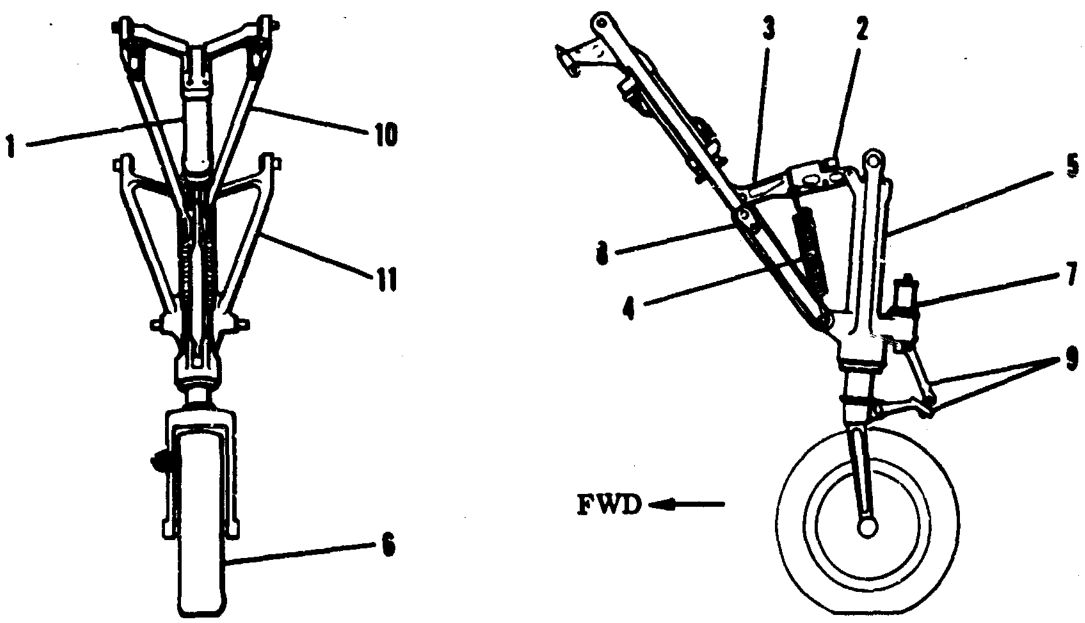


FRONT VIEW

SIDE VIEW

NOSE LANDING GEAR

Figure 56. Aircraft Landing Gear Configuration.

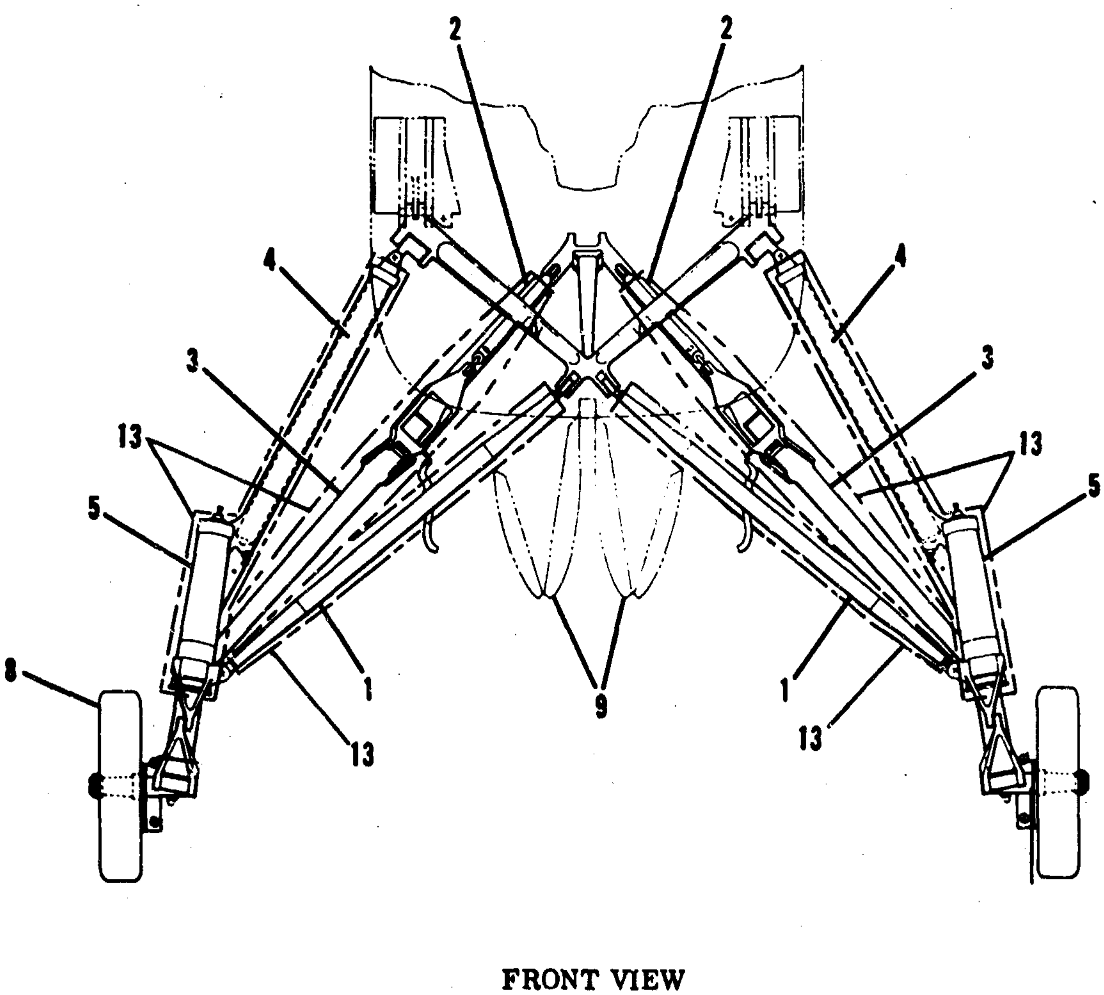
A

OSE LANDING GEAR

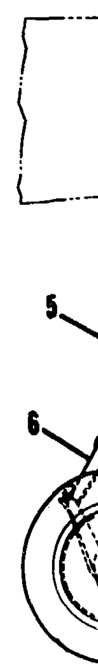
.G HYDRAULIC ACTUATOR
.G LOCKED SWITCH
RY BRACE
RING
OCK STRUT
.G WHEEL ASSEMBLY
IMMY DAMPER
OUND LOCK PIN
RQUE LINKS
PPER DRAG BRACE
WER DRAG BRACE

MAIN LANDING GEAR

1. SIDE SWAY BRACE
2. MLG HYDRAULIC ACTUATOR
3. DRAG STRUT ASSEMBLY
4. EMERGENCY PNEUMATIC SYSTEM RESERVOIR
5. SHOCK STRUT
6. TORQUE LINK
7. MLG FOLD MECHANISM
8. MLG WHEEL AND BRAKE AS
9. DOORS (SHOWN IN "OPEN" P
10. DOOR MECHANISM
11. UPLATCH
12. 2-POSITION MECHANISM (MO
13. INSULATION



FRONT VIEW



CTOL
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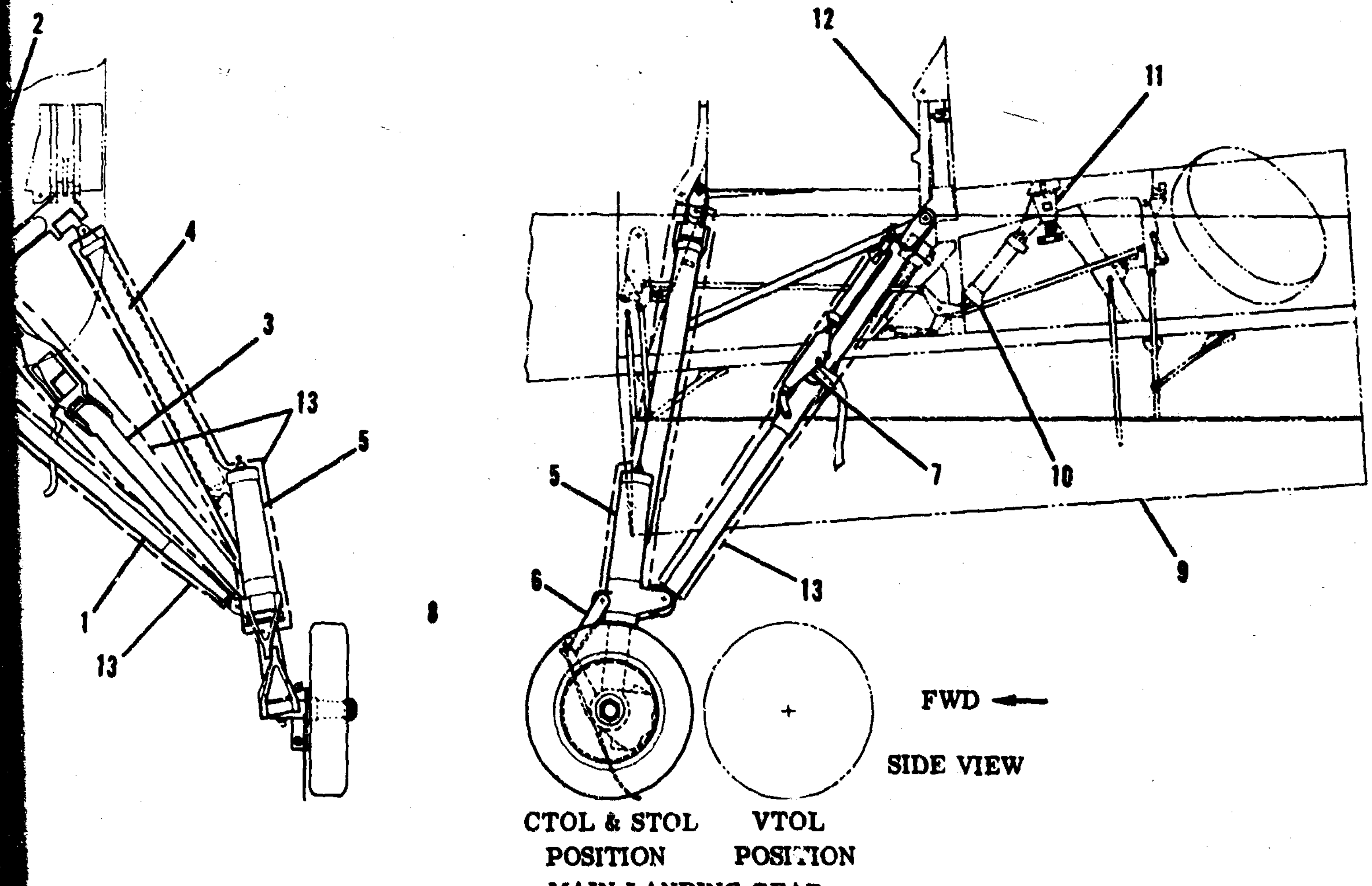
MAIN LANDING GEAR

1. SIDE SWAY BRACE
2. MLG HYDRAULIC ACTUATOR
3. DRAG STRUT ASSEMBLY
4. EMERGENCY PNEUMATIC SYSTEM RESERVOIR
5. SHOCK STRUT
6. TORQUE LINK
7. MLG FOLD MECHANISM
8. MLG WHEEL AND BRAKE ASSEMBLY
9. DOORS (SHOWN IN "OPEN" POSITION)
10. DOOR MECHANISM
11. UPLATCH
12. 2-POSITION MECHANISM (MODE CHANGER)
13. INSULATION



MAIN LANDING GEAR

C

utilizes high-viscosity silicone fluid as the damping medium. The nose gear and damper have been drum-tested at taxi speeds of up to 145 mph over the full load range. The 8-by-4.4 tubeless tire has a 10-ply rating and an inflation pressure of 185 psig.

Landing Gear Control System

The landing gear control handle (on the left-hand side of the main instrument panel) operates an electrical command circuit controlling four solenoid-operated, four-way, open-center selector valves. Separate valves are used to control the nose gear, the main gear, the main-gear-uplatch two-position mechanism, and the main gear door mechanism. The gears, doors, and mode changer are interlocked and sequenced by limit switches, which also provide position logic for the cockpit signal system. The landing gear operation is powered by the aircraft No. 1 hydraulic system.

Inadvertent retraction while the aircraft is on the ground is prevented by limit switches mounted on the torque links of each main gear. These switches deenergize a solenoid-operated pawl, which, in the deenergized position, locks the pilot's control handle in the gear-down position. In the event of an impending collision or a similar dire emergency, the pilot can elect to belly the aircraft. The down-lock release button overrides the solenoid pawl lock. If the pilot holds the button and simultaneously selects the gear-up position, the gear will retract. Four circuit breakers (for landing gear control, indicators, warning, and air circuits) are located on the circuit-breaker panel to the right of the pilot's seat.

Brake System

Single-disk brakes are provided on each main wheel. They are self-adjusting. The hydraulic brakes are manually operated by toe-type rudder pedals linked to separate master cylinders for each brake. The brake hydraulic system is independent of the aircraft hydraulic systems. Antiskid devices and a parking lock are not provided.

Cockpit Signal System

The landing gear cockpit signal system utilizes five visual signals and an auditory signal. Three mechanical visual signals, one for each wheel, are located on the left console panel. Each displays a wheel if the wheel is down and locked, stripes if a wheel is in transit (or not in commanded position), and the word UP if the wheel is up and locked and the door is closed and locked. An amber light on the left console panel indicates that the STOL gear position has been selected; the main gear wheels will remain in the forward (jet-mode) position during both fan-mode and jet-mode configurations. A red light in the gear control handle illuminates and a

headset auditory warning starts if speed drops below 150 knots at any altitude. The auditory warning may be silenced by depressing the SILENCE button, which is to the left of the landing gear control handle. If the aircraft descends to an altitude below 3300 feet (absolute), the auditory warning will restart and may be resilenced. In this event, the auditory warning switch will reset if the aircraft subsequently climbs to 4300 feet (absolute). The red light will extinguish when the drag struts lock all three gears in the extended position. It will light after retraction if the gear fails to lock up completely or if the main gear doors are not locked in the closed position.

Emergency Pneumatic System

During emergency operations, the landing gear indications and warnings will be the same as during normal operations, provided that malfunction does not affect (1) the warning circuitry and (2) the mode-change system. The emergency system will operate even if a 100-percent electrical failure occurs. In this event, the pilot is dependent on a ground or chase plane observer for confirmation of gear position.

The emergency system (see Figure 57) does not operate the two-position mode-change mechanism. The gear position indicator displays wheels only if the gear is down and locked in the mode commanded. If the aircraft is in jet mode or if STOL override is selected and if the mode-change mechanism is disabled, the main gear indicators will display stripes and the main wheels cannot move to the forward wheel position. However, if the red light in the gear control handle goes out, the main gears are down and locked, and an emergency conventional landing may be made with the wheels in the aft (fan) position. This is classed as an emergency landing, and structural inspection of the main gear area will be required prior to the next flight.

Emergency extension of the landing gear is accomplished by use of a 1-shot pneumatic system (see Figure 57) in the event that pressure is lost in hydraulic system No. 1 or the electrical command circuitry malfunctions. Gaseous nitrogen at 3000 psi is stored in the upper ends of each main landing gear shock strut leg. The total reservoir capacity is 410 cubic inches. The charging valve and pressure gage are located in the main landing gear bay. Operations during an emergency extension proceed as follows:

1. DOWN is selected with the cockpit control handle.

2. The T-handle on the cockpit floor to the left of the pilot's seat is pulled up (a pneumatic pressure switch deactivates the normal command circuit).

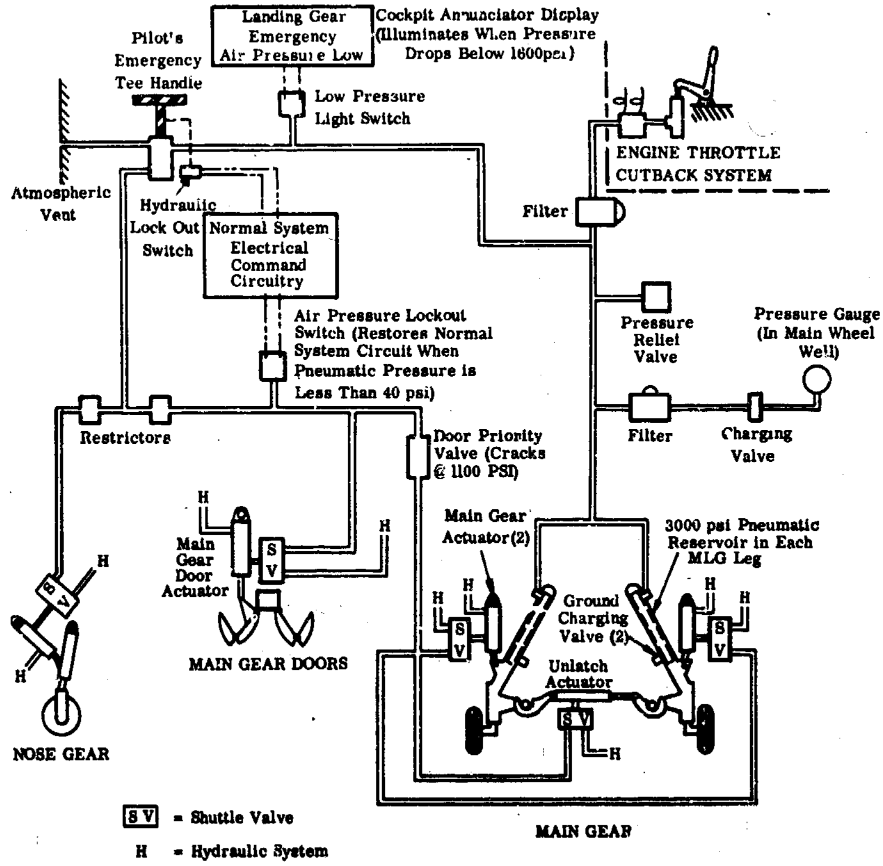


Figure 57. Emergency Pneumatic System Schematic.

3. Mode-change, indicator, and warning circuits remain active (unless disabled by the aircraft malfunction).

4. Pneumatic pressure will first be applied to the nose gear and to the main landing gear doors. When the main gear doors have opened, a priority sequence valve cracks after pressure has built up to 1100 psi; pressure is then applied to release the main gear uplatches and to drive the main gears to the down and locked position.

The "emergency landing gear, pressure low" warning on the cockpit annunciator panel illuminates when the stored system pressure drops to 1650 psi. This is the minimum pressure required to ensure satisfactory operation of the emergency system.

The approximate extension time from pilot command to the time when the gears are down and locked (in fan-mode configuration) is 10 seconds at 3000 psi (pneumatic pressure) and 30 seconds at 1650 psi (pneumatic pressure).

MAINTAINABILITY HISTORY AND RECOMMENDATIONS

Failure Rate

Eighteen landing gear system installation failures were reported. Of these, two were reported during a flight or ground test mission. The man-hours expended in correcting discrepancies are shown in Table XIII.

Stability During Taxiing and Hovering

A low margin of ground stability during taxiing is inherent with the narrow-track fuselage gear. This is alleviated somewhat by the use of differential braked steering. Instability under the lift-system forces and dynamic ground reactions has been encountered during hovering operation; the resulting scrubbing action causes wear on the tires.

Modification to Insulation

In its present location, the main landing gear requires heavy insulation blankets to protect it against adverse thermal and high-velocity gas impingement from the wing-fan exhaust.

The present installation of this insulation prevents easy access for maintenance work in certain areas, among which are the following: electrical harnesses to landing gear ground-actuated switches; brake and pneumatic plumbing; and shock strut air reservoir charging and drain plugs.

Redesign of the insulation blankets to provide easy removal and installation will greatly improve inspection access and maintainability of the system. In addition, derivative aircraft with a similar configuration should consider the use of high-temperature materials in landing gear components.

Nonstandard Jacking Provisions

Two aircraft jack points are provided on the wing front spar at buttock line 100.75 and one on the fuselage at station 276. Jack points are also provided at the main gear wheels, but they are not compatible with standard service jack equipment. No jacking provision is made on the nose gear. To remove a main wheel, it is necessary to jack the aircraft on the wing and fuselage pads. It has also been noted that installation of the nose gear towbar can damage the axle nut and cotter key.

Nonstandard jacking provisions, *per se*, are not detrimental to continued successful operation of the XV-5A, but the existing situation would be unacceptable on production-type aircraft.

TABLE XIII. MAN-HOURS EXPENDED IN CORRECTING DISCREPANCIES IN LANDING GEAR

Part Name	Preparation	Diagnosis	Accessibility	Corrective Action	Reassembly	Check-out	Total Man-Hours	Remarks
Main Gear Wheel Assembly	0.2	0.1	0.3	4.1	0.3	0	5.0	Wheel removed
Main Landing Wheel Key	0.2	0.1	0.3	2.0	0.3	0	2.9	Wheel removed
Landing Gear Strut	0.1	0.5	4.0/2.0*	1.0	4.0/2.0	0.5	10.1	O-ring replaced
Pneumatic Priority Valve	0.1	0.1	0.1	0.5	0.1	0	0.9	Replaced
Door Lock Switch, Main Landing Gear	0.2	0.1	1.5	6.0	3.5	0.8	12.1	Replaced - wire showed signs of excessive heat
Down Lock Switch, Main Landing Gear	0.2	0.1	1.0	4.0	2.0	0.8	8.1	Replaced - wire showed signs of excessive heat
Tire	0.5	0.1	0.1	0.2	1.0/0.0	0	1.9	Replaced - worn
Tire	0.5	0.1	0.1	0.2	1.0/0.0	0	1.9	Replaced - worn
L.H. and R.H. Hose Assembly, Main Landing Gear Emergency Air	0.2	0	0.1	1.0	0.1	0.6	2.0	Replaced - swaged fittings leaking due to exposure to heat
Tube Assembly - Emergency Air (routed under crossover duct)	0.2	0.2	1.0	2.6	1.0	0.1	5.1	Replaced - was ruptured
Nose Tire	0.2	0	0.1	1.0	0.2	0.1	1.6	Replaced - 1/4-inch cut
Switch Shaft Main Gear Retraction Cylinder Assembly	0.2	1.0	0.3	1.0	0.3	0.2	3.0	Shaft lubricated - was tight in bushing
Indicator, Landing Gear	0.2	0.1	0	0.1	0	0.1	0.5	Replaced
R.H. Tubeless Tire	0.2	0	0.1	0.2	1.5	0.1	2.1	Replaced - 6-inch radial cut

*If 2 numbers are divided by a slash, the first number represents the man-hours; the second, the clock hours.

Brake System History

Following is an account of the brake system performance:

1. No brake structural failures or fires.

2. No complaints of runaway brake pedals or master cylinder overflow.

3. System difficult to bleed properly.

4. Heavy pedal forces and unresponsive brake action.

5. No emergency straight-line stops.

6. Two failures of brake disk drive keys. (The keys are steel inserts around the inside of the wheel rim which key the brake disk to the wheel. The failures are probably due to the heavier disk now used. The manufacturer has not confirmed this opinion, although larger retaining screws have been installed in one set of wheels.)

The troubles that were encountered were attributed to the following:

1. High brake usage was required to keep the aircraft down to a safe taxi speed because of the high installed residual thrust of the jet engines.

2. Light brake disks rapidly reached temperatures at which the lining friction coefficient deteriorated rapidly, thus requiring the pilot to use extreme pedal force, which resulted in high system pressures.

3. Magnesium casting housing strength was seriously reduced because of overheating.

4. Disk warpage caused a redistribution of bending moment on the anvil portion of the castings.

5. Heavy pedal pressures and resulting casting deflections caused the automatic adjustment feature in the brake mechanism to overcompensate for wear, which resulted in dragging brakes.

6. Rough machined surfaces in the bore of the cockpit master cylinders caused erratic operation of the valve mechanism in the cylinder.

7. The existing cockpit pedal geometry provides a rapid increase of mechanical advantage as the pedal angle increases. This permits the pilot to exceed the recommended lining contact pressure. If the linings are already at their maximum operating temperature, this could contribute to rapid failure of the linings.

The following corrective actions were taken:

1. Changed casting material from magnesium to aluminum to provide increased stiffness and strength at elevated temperatures.

2. Increased brake disk thickness from .312 inch to .468 inch to provide increased heat sink capacity and higher resistance to warpage and dishing.

3. Revised initial setup of automatic adjusters to maintain disk/lining clearance after application of heavy pedal pressures.

4. Honed out brake master cylinder bores to a finer finish. Also added an overflow header tank to collect cockpit spillage.

5. Instituted use of thrust spoilers as standard operating procedures during taxi operations.

6. Made a flight test evaluation of brake lining materials. This indicated that an improvement in lining life can be expected by the adoption of friction-type material.

The brake situation is perhaps a classic example of the conflict between VTOL and CTOL requirements. The main difficulties emanate from the following requirements of the original specifications:

1. Light weight.

2. Short stopping distance (approximately 1500 feet) in the CTOL mode.

3. Simple maintenance.

4. Limited number of stops (10 required).

5. Pedal pressures during brake landing roll in accordance with MIL-B-8584b.

6. Static torque sufficient to hold full engine thrust in accordance with MIL-B-8584b.

The brake system is not adequate for operating a high-residual-thrust vehicle over a vast area such as that of Edwards Air Force Base, when the brakes are also relied upon to steer the vehicle. The brake capacity requirements for operational- and trainer-type aircraft have been reviewed in light of the type of operation at Edwards Air Force Base. The most attractive approach points to a single-disk brake, built largely from the existing components. A thicker disk (5/8 inch) would be used in conjunction with two of the existing cylinder housings. The lining area would be doubled. The cockpit linkage and master cylinder would have to be revised. As noted above, some revision to the linkage would be desirable even with the present brake. Use of dual housing would permit a dual brake system on each side of the aircraft. A single-line failure would then cause only a 50-percent loss of braking. At present, a single-line failure will result in a 100-percent loss of braking on one side, which is undesirable.

A fully tactical type aircraft will probably require a full-powered antiskid-type brake system, employed in conjunction with nose-wheel steering for maximum-effort STOL landings.

The use of a brake parachute offers a solution to the conflict between braking requirements for heavily loaded STOL/CTOL missions and less critical VTOL requirements. A review should be made of the weight trade-off between brakes and drag chute installations for follow-on aircraft.

DESIRABLE FEATURES

1. The single-disk brakes are self-cleaning and self-adjusting and permit easy inspection and rapid, economical relining.

2. The forward-retracting nose gear provides free-fall capability for emergency extension.

3. The nose-gear jury strut locks in both extended and retracted positions.

4. The nose-gear doors operate directly by linkage to the retracting mechanism.

5. No electrical or hydraulic system power is needed for emergency gear extension.

UNDESIRABLE FEATURES

1. Narrow-track gear contributing to reduced ground handling stability during hovering lift-off and touchdown and during taxiing.

2. Main gear insulation installation, accessibility, and maintenance.

3. Down-lock override and retraction sequence-timing relationships (including main landing gear retraction time).

4. Emergency pneumatic system power utilization.

5. Wheel and tire maintenance provisions.

6. Nose-gear towbar installation.

7. Maintainability and reliability of flush-type lubrication fitting.

8. No parking brake.

9. Only one of two hydraulic systems used for retracting or extending the landing gear.

RECOMMENDATIONS AND SUGGESTIONS FOR UNEVALUATED IMPROVEMENTS

Retraction of Landing Gear

The existing landing gear system configuration with down-lock-override capability introduces an undesirable condition because of the 4-second nose-gear retraction time versus the 12-second main-gear retraction time. The suggested corrective action is (1) to adjust both gear retraction restrictors to provide the same retraction time or (2) to remove the override capability. It is recommended that this problem be evaluated and that appropriate corrective action be taken at the earliest possible time. This evaluation should include consideration of the following normal gear retraction sequences:

1. First, the nose-gear toggle lock breaks; then, the nose gear retracts.

2. If the main gear is in jet-mode position, it must first cycle to fan-mode position before the retracting actuators unlock; next, the drag link toggle breaks; then, the main landing gear retracts. (Position transfer presently requires approximately 7 seconds.)

Landing Gear Extension

The main landing gear extends forward against the slipstream. Therefore, some system is mandatory to assist the gear in falling to the extended position. The emergency pneumatic system, which is completely divorced from all hydraulic and electrical circuitry (except for cockpit indication), is satisfactory for continued operation of the XV-5A. Use of the upper portion of the main struts as storage reservoirs is a worthwhile weight- and space-saving feature. The use of the emergency system to power the throttle cutback system is undesirable. In the present installation, a leak in the throttle cutback system or emergency landing gear system would cause complete loss of emergency system pressure and, consequently, loss of both functions. Revised plumbing should be investigated to determine if it is possible to reduce the likelihood of loss of either or both functions. (Reference also the throttle cutback system discussion in the flight controls section of the report on page 108.) It has been suggested that the emergency pneumatic system could be used to power the antispin/drag parachute deployment system to improve parachute system reliability. This suggestion has not been evaluated. It should be included in the study of the emergency pneumatic system and the throttle cutback system.

Wide-Track, Wing-Mounted Gear

A wide-track, wing-mounted gear would offer the following advantages:

1. Ground stability would be improved and thus the possibility of tip-over would be lessened.

2. Adequate differential braked steering would be easy to achieve.

3. With the gear located outboard of the fan installation, relief from fan-exhaust thermal, blast, and lift-loss effects would be provided.

4. The requirement for a two-position gear would be eliminated, thus simplifying retraction and at the same time improving system reliability by reducing system complexity.

However, the radical increase of track dictated by the wing-fan cutout may have the following disadvantages:

1. The severe ground-looping tendency induced by momentary loss of wheel adhesion on one side could be a real problem. An antiskid system to cope with different runway conditions existing at the widely separated pairs of wheels would have to be considered.

An antiskid system and nose-wheel steering will almost certainly be required to stop an operational-type airplane in the shortest distance, particularly on a rough field. The wheels will necessarily be at the point of skidding, and directional control by nose-wheel steering will be essential.

2. The lateral moment of inertia would be increased.

3. Long plumbing and wiring runs would be required.

4. Large gear stowage pods would be required on the wing.

Flush-Type Lubrication Fittings

It is recommended that flush-type lubrication fittings, which are presently used in some joints in the nose and main gear, be avoided wherever possible in future designs because they are easy to overlook, are difficult to lubricate, and can blow out under pressure.

ORGANIZATIONAL LEVEL MAINTENANCE

TRAINING OF MAINTENANCE TECHNICIANS

The maintenance technicians presently in the military organizational structure can perform the tasks required after receiving proper training. So that the maintenance technicians can learn more about the systems and the maintenance requirements on a V/STOL concept similar to the XV-5A, it is recommended that the following areas be stressed in the training of maintenance personnel:

1. The stability augmentation system.

2. The electrical and mechanical flight control inputs to the mixer boxes in the flight control system, to include the difference in the controls when the control inputs are phased in or out during hovering-, conventional-, and conversion-type flying.

3. The propulsion system that provides vertical lift.

4. The effect of a high-temperature (170°- 730°F) environment on the performance of aircraft systems, subsystems, and components; the high temperatures are caused by hot exhaust gases from the jet engines passing through hot-air ducts to the nose, by heat radiating from the wing-fan turbines, or sometimes by leakage at duct joints and cracks.

MAINTENANCE MAN-HOURS EXPENDED FOR TIME-COMPLIANCE INSPECTIONS

Incorporation of a number of suggestions for improving maintainability of the XV-5A, together with increased maintenance know-how acquired during this evaluation program, led to a determination of man-hours required to perform organizational level maintenance. The requirements are shown in Table XIV. Although the man-hours are high, they are tolerable when it is considered that the design of the aircraft was a new concept and that it was fabricated for research work only.

The following facts should be considered when the table is read:

1. The table is based on two 30-minute flights per working day.

2. Supervision, technical support, logistics support, and instrumentation support are not included.

3. The table does not include man-hours for unscheduled maintenance and repairs.

4. The manpower expended for turnaround inspection includes refueling the aircraft.

5. Functional inspection (required to check out redundant aircraft systems) was good for a period of 48 hours.

6. The 10-hour inspection includes man-hours for performing the 5-hour inspection; the 30-hour inspection includes man-hours for performing the 5- and 10-hour inspections.

TABLE XIV. MAN-HOURS REQUIRED FOR ORGANIZATIONAL LEVEL MAINTENANCE

Type of Inspection	Men Required	Clock Hours	Man-Hours
Preflight	4	3	12
Turnaround	5	1	5
Postflight	3	2	6
Functional	4	2	8
5-Hour	3	5	15
10-Hour	5	26	130
30-Hour	5	64	320

RECOMMENDED INSPECTIONS AND MAINTENANCE MAN-HOURS FOR PRODUCTION AIRCRAFT

Recommended types of inspections and organizational maintenance man-hour estimates for conducting time-compliance inspections on a production lift-fan aircraft similar to the XV-5A, under ideal working conditions, are

shown in Table XV. Ordnance and electronic equipment, depending on types, would require additional maintenance man-hours; and it is estimated from past experience that unscheduled maintenance would require 4 maintenance man-hours per flight hour during the first year of operation.

TABLE XV. RECOMMENDED MAN-HOURS FOR ORGANIZATIONAL LEVEL MAINTENANCE

Type of Inspection	Men Required	Clock Hours	Man-Hours
Preflight	1	.5	.5
Turnaround	2	.5	1.0
Postflight	2	1.0	2.0
25-Hour	3	8.0	24.0
50-Hour	5	8.0	40.0
100-Hour	5	16.0	80.0
300-Hour	5	40.0	200.0

SPARE PARTS USAGE

Table XVI lists the spare parts that were used for the propulsion, control, electrical, hydraulic and pneumatic, and landing gear systems; the aircraft instruments and miscellaneous spare parts are also listed.

TABLE XVI. SPARE PARTS USAGE		
Description	Part No.	Quantity Used
PROPULSION SYSTEM		
Fuel Boost Pump	SCDP0029	1
Rod End	8-10609-3	24
Rod End	8-10609	9
Rod End	2BREM-LHS-4A	1
Blade Platform	4012001-159G2	35
Blade Platform	4012001-164G2	35
Drive Shaft	SCDP0021	1
Diverter Valve Switch	12HRIZ-RB	1
Valve	SCDP0051	5
Valve	SCDP0030-1	1
Fire Extinguisher	SCDP0043-1	2
Engine Mount Blank	143P005-7	1
Engine Mount Blank	143P005-9	1
Fuel Shutoff Valve	AV24B1108	1
Rod End	8-10609-5	1
Valve	SCDP0012	1
CONTROL SYSTEM		
Throttle Brake	SCDK0011-1	1
Monoball Bearing	BAR-8480	1
ELECTRICAL SYSTEM		
Generator Control Panel	3S2060DC125A1A	3
Inverter	32B50-4-B	7
Flap Motor	SCDE0039-1	2
Actuator, Vector	SCDE0045-1	6
Actuator, Roll Trim	SCDE0044-1	1
Switch, Actuator	JE-61	1
Generator	2CM299D1	4
Silver Zinc Cells for Batteries	Type S-25	25
Audible Warning Generator	ALL-0543	3
Antenna	AT-256 A/ARC	1
Radio	ARC 51X	3
Wing-Fan Door Latch Actuators	SCDE0028-1	3
Electric Mixer	143E012-1	1
Switch Actuator	JV82	1

TABLE XVI. - Continued

Description	Part No.	Quantity Used
HYDRAULIC AND PNEUMATIC SYSTEM		
Pitch-Fan Door Servo	SCDH000-1	1
Horizontal Stabilizer Actuator	SCDH0008-1A	2
Wing-Fan Door Actuator	SCDH0009-1	3
Hydraulic Pump	SCDH0007	2
Wing-Fan Door Actuator	SCDH0009-2	3
Wing-Fan Door Actuator	SCDH0009-101	2
Wing-Fan Door Actuator	SCDH0009-102	2
Wing-Fan Louver Actuator	SCDH0002-101	1
Restrictor	SCDH0012-33	2
Restrictor	SCDH0012-20	1
Actuator	1510L300-2	1
LANDING GEAR SYSTEM		
Main Landing Gear Tire	20 x 4. 4	9
Damper	8-14200	1
Brake Assembly	SCDL0003-13	5
Brake Disk	SCDL0003-15	4
Shimmy Damper	1511L400	2
Brake Liner	SCDL0003-17	27
Brake Disk	SCDL0003-9	1
Nose Wheel	SCDL0004	1
Inner Race	3010-329	1
Main Landing Gear Wheel	SCDL0003-11	3
Nose Wheel Tire	18 x 4. 4	1
Landing Gear Quadrant Control	A-50M3-1	1
AIRCRAFT INSTRUMENTS		
Low Airspeed Indicator	586BK-0155	2
Bellows	G404-50	4
Vector Angle Indicator	8DJ50MAP-2	3
Fuel Flow Indicator	6680L042550	1
Transmitter	6685L087924	1
Indicator	6680L0425502805	2
Pressure Gage	2725-718	1
Pressure Gage	6901-714	2

TABLE XVI - Continued		
Description	Part No.	Quantity Used
Sterer Valve	27900	4
Pressure Transmitter	46139-G20-50	4
Hydraulic Pressure Indicator	SRD-7K	1
Pressure Transmitter	MS28005-5	2
Temperature Indicator	6620-531-4630	1
Attitude Indicator	J8	1
Indicator	AR100-103	2
Waugh Box	AC-106	3
Card	2-17744A	3
Card	2-1850	2
Rate of Climb Indicator	AN5825-1	1
Rate of Climb Indicator	AN5825-7	1
Indicator	AN5536-2A	1
Indicator	AN5839-2	1
MISCELLANEOUS		
Oxygen Filler Valve	ZV861B	2
Valve	27900	1
Valve	AVLF-1166	2
Tail-Cone Cap	143F163-1	2
Valve	MS28889-1	2
Drain Valve	MS29530-6	3
Selector Valve	362-0371	1
Valve	707315	1
Bearing	MK54A	1
Bushing	DBS-030	3
Switch	4TL1-2D	1
Counter	159104	1
Valve	A63007	1
Bushing	DBA-8-030	14
Streamer Assembly	NAS 1091-30H	2
Switch	402EN1-6	3
Link	4012153-359P2	1
Bearing	BAN7325	2
Valve	13530	1

CONCLUSIONS

It is concluded that:

1. During fan-mode operation, the high-temperature environment in the XV-5A was the primary cause of malfunctions and numerous fatigue failures in aircraft systems and components that are located near the hot gas ducting.

2. The excessive time (in man-hours) required to keep the XV-5A in an airworthy condition is attributed to the lack of maintainability design effort.

3. The thrust-stand tests were more detrimental to the aircraft than any other tests conducted during the program.

4. The design refinements that will be required to build this concept into an operational model are not beyond the engineering technology available during the 1967-1971 time period.

RECOMMENDATIONS

It is recommended that:

1. Increased design emphasis be directed toward reducing the high-temperature environment either by means of a more effective cooling system in the aircraft or by designing a duct that will transfer hot gases from 1700° to 1800°F at 75 psi to prevent degradation of the structure, systems, subsystems, and components. If a new duct is designed, it must also be easy to install and inspect.

2. When a research aircraft of a new concept is being designed, the design engineering philosophy be such that standard, proven systems and off-the-shelf components are given maximum consideration so as to decrease engineering and maintenance costs.

3. When more than one research aircraft of the same concept is constructed, the first aircraft fabricated be used for full-scale wind tunnel and/or static thrust-stand tests only.

4. Additional design studies be conducted to advance the XV-5A-type concept.

5. More emphasis and requirements be placed on design, maintainability, and reliability considerations in future research aircraft contracts.

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13. ABSTRACT

In the past, little formal effort has been expended by the U. S. Army in evaluating the maintenance and systems aspects of experimental aircraft.

The data compiled during this evaluation were used to determine the effectiveness of design as it applies to maintainability of the overall aircraft, its systems, and its subsystems and, in cases of deficiencies, to recommend improvements and to specify areas that require further research before derivative XV-5A-type aircraft are constructed.

Each problem area was analyzed to determine whether the discrepancies resulted from the austere research aircraft program or whether they were inherent in the lift-fan concept. Results of this study uncovered the desirable and undesirable features of 10 of the XV-5A aircraft systems.

Design refinements that will be required to build the lift-fan concept into an operational model are not beyond the engineering technology available during the 1967-1971 time period.

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